PERFORMANCE SUMMARY
FOR THE

IRIS

SOUNDING ROCKET VEHICLE

REPORT NO. AST/E1R-13324

18 April 1961

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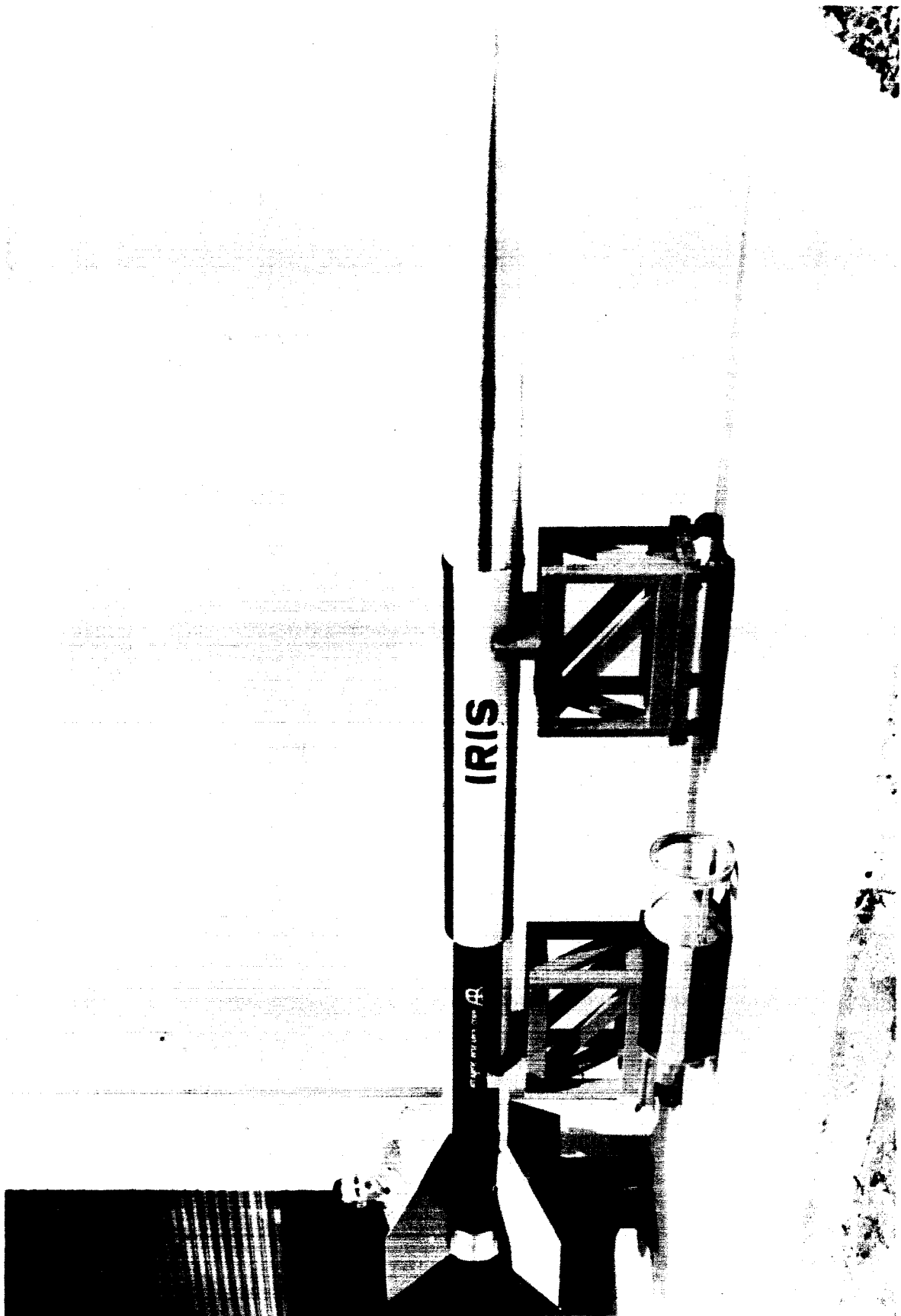

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IRIS

INTRODUCTION

This Sounding Rocket Handbook report is one of a series prepared by Vought Astronautics, a Division of Chance Vought Corporation, for the National Aeronautics and Space Administration under Contract No. NAS1-1013. This contract was administered by the Langley Research Center under the technical direction of Hal T. Baber, Jr., of the Vehicle Performance Branch, Applied Materials and Physics Division, Langley Research Center. This report presents data for one of the eighteen vehicle systems listed below:

Vehicle	Handbook No.	Vehicle	Handbook No.
Aerobee 100	AST/E1R-13318	Journeyman	AST/E1R-13327
Aerobee 150A	AST/E1R-13319	Journeyman B	AST/E1R-13328
Aerobee 300A	AST/E1R-13320	Jaguar	AST/E1R-13329
Arcas	AST/E1R-13321	Little Joe	AST/E1R-13330
Arcon	AST/E1R-13322	Nike-Asp	AST/E1R-13331
Exos	AST/E1R-13323	Nike-Cajun	AST/E1R-13332
Iris	AST/E1R-13324	Shotput	AST/E1R-13333
Jason	AST/E1R-13325	Skylark	AST//1R-13334
Javelin	AST/E1R-13326	Strongarm	AST/E1R-13335

In addition to the handbooks on each vehicle, the following handbooks have been prepared:

Handbook	Handbook Number
Summary Report	AST/E1R-13337
Rocket Motor Ballistic Data Report	AST/E1R-13336 (Confidential)
Cost and Reliability Summary	AST/E1R-13338

FOREWORD

The primary purpose of this report is to aid in the preliminary selection of a vehicle for a specific payload mission. Performance data in this report show a broad flight regime and have not been modified by restraint items such as aerodynamic heating, range safety and other detail factors. In fact, this cannot be done until a mission has been established. Thus, caution must be used in extracting detail data. It is believed that the information presented will allow the user to consider all the major aspects of the booster system, and will serve as a guide in payload system integration.

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LIST OF FIGURES (Continued)

VEHICLE DESCRIPTION

Summary

NAME OF VEHICLE	IRIS
DESIGNATION	52 KS 3850, MARC 13A 1
MANUFACTURER	ATLANTIC RESEARCH CORPORATION ALEXANDRIA, VIRGINIA
NUMBER OF STAGES	2
LAUNCH WEIGHT (NO PAYLOAD)	1419.0 POUNDS
OVER-ALL LENGTH	278.7 INCHES
MAXIMUM DIAMETER	12.75 INCHES
PRIME USER	NASA
NET PAYLOAD	
NOMINAL	100 POUNDS
MINIMUM	75 POUNDS
MAXIMUM	200 POUNDS
VOLUME	4.5 CU. FT. (EXCLUDING EXTENSION)
PERFORMANCE AT NOMINAL NET PAYLOAD APOGEE ALTITUDE (VERTICAL LAUNCH)	175 NAUTICAL MILES
ACCELERATION, MAXIMUM	13.9 "g"

The vehicle assembly and staging weight is shown in Figure 1. Figure 2 shows the aerodynamic outline of the payload compartment used in the analysis and the associated usable volume.

Background

Design studies and early work on Iris motor development were initiated for the Naval Research Laboratory by the Atlantic Research Corporation in 1956. Motor and vehicle development were completed for NASA's Goddard Space Flight Center in 1960 and included a six-round motor qualification testing program and detailed aerodynamic design. Three performance flight tests were conducted by Goddard in late 1960 and early 1961. Iris is a two-stage, solid propellant rocket designed to be launched from a 160 foot, 4 rail tower of the type installed at Wallops

Island, Virginia. Step one for the Iris is a separate thrust unit consisting of seven 4 inch diameter rocket motors which is used to increase exit velocity from the tower. It is not mechanically attached to the IRIS and falls away after the vehicle exits from the tower. Thus, stage one operation utilizes both step one and step two thrust, while stage two requires only the second step motor. The ignition system incorporates a relay with a built-in time delay to insure that step one ignites at launch before step two does.

IRIS

DIMENSION INCHES   DIMENSION METERS

Sta. 0.0

PAYLOAD
COMPARTMENT

12.5 →   ← .32

12.0 →   ← .30

238.1   6.05

278.7   7.08

159.1   4.04

← * →

IRIS
(52 - KS - 3850)

45°

36   .91

19   .38

IRIS BOOSTER
(.8KS 18800, MARC.14 B1)

12.75 →← .32

47.14 →← 1.20
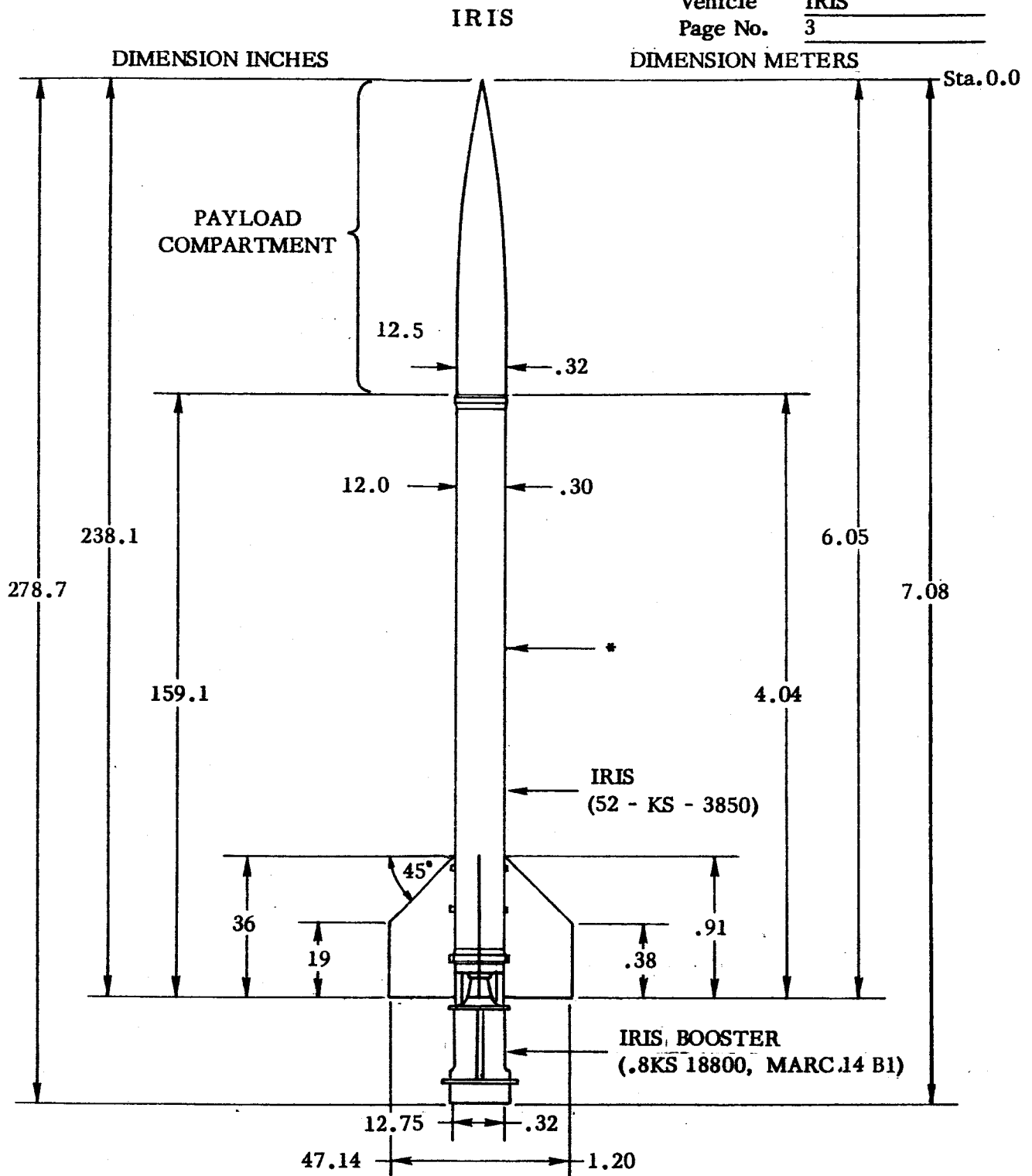
PERFORMANCE WEIGHTS

POUNDS	LESS PAYLOAD	KILOGRAMS
1419.0	Launch	643.6
1336.3	B. O. 1st Stage	606.1
1204.8	Drop 1st Step	546.5
247.0	B. O. 2nd Stage	112.0

* Aerodynamic Reference Area = .886 ft^2

FIGURE 1 VEHICLE ASSEMBLY AND STAGING WEIGHTS

DIMENSION INCHES   DIMENSION METERS

80   2.03

                                          — Nominal Nose Cone

20   .51
                                          — Extension
                                          (OPTIONAL - NOT USED
                                          IN THIS REPORT)

12.5   .32
Dia.
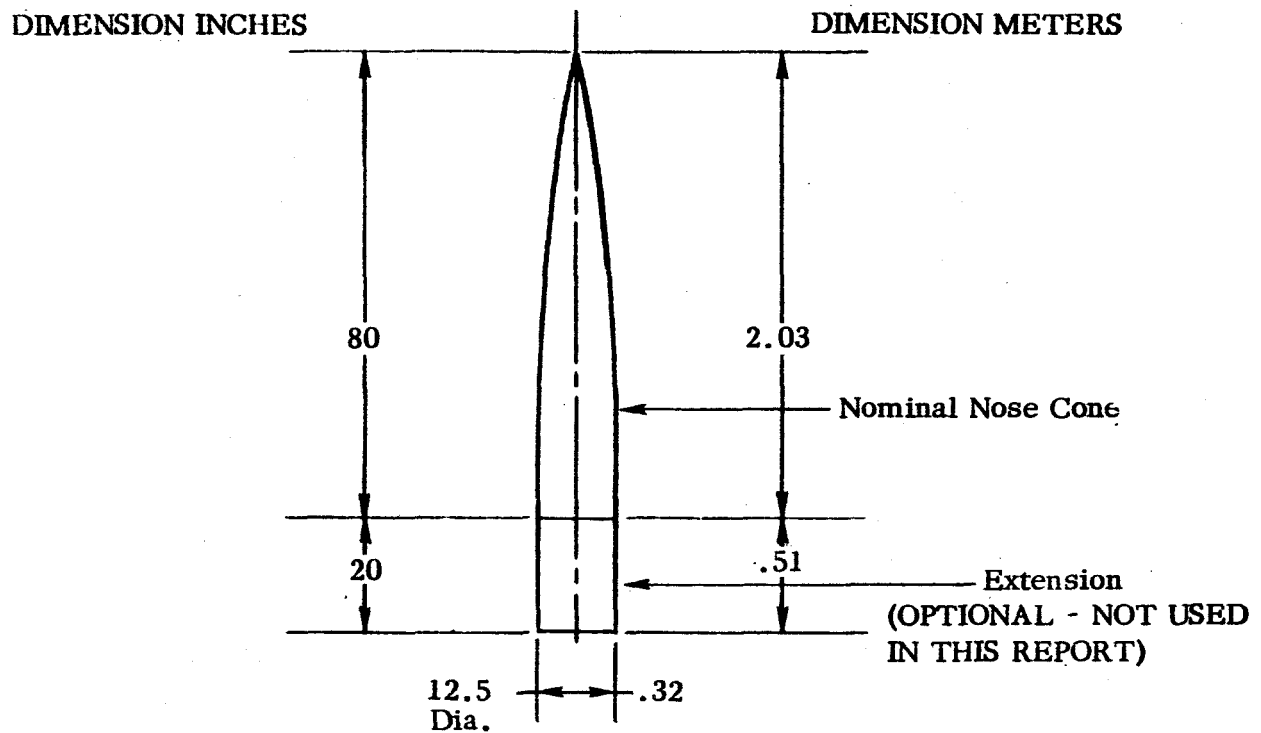
NOMINAL AVAILABLE PAYLOAD VOLUME = 4.5 CU. FT. (.129 CU. METERS)
WITH 20" CYLINDRICAL EXTENSION = 5.8 CU. FT. (.166 CU. METERS)

FIGURE 2 PAYLOAD COMPARTMENT

FLIGHT PERFORMANCE

The flight performance data presented in this study show a very broad flight regime for each vehicle. Modifications to the data have not been made to account for factors such as the launch site, launcher elevation limits, range safety, and vehicle-payload environment. Consideration of these factors usually results in limitations being placed on the flight regime. Some limitations may be removed by minor modifications, while in the case of range safety, the limitation may be revised with no modification as the vehicle builds a good operational history. If the flight performance data were based on a set of firm limits, it would be very difficult to extrapolate the data. However, it is rather easy to restrict, when necessary, the broad flight regime shown in figures 3 through 16. Some degree of caution must be exercised in interpreting these figures. For example, the vehicle was considered to be a "clean" aerodynamic configuration, i.e., it was assumed to have no external antennae, even though certain experiments in the past have been flown with antennae. Further, all performance is presented for net payload, as defined in the Nomenclature.

Flight performance calculations were conducted with an IBM 704 digital computer using two degree of freedom analysis on a spherical, non-rotating earth. The routine considered aerodynamic coefficients to be Mach number-dependent, while thrust was computed by correcting time-dependent vacuum thrust for ambient pressure. The 1959 ARDC model atmosphere was used.

Trajectory

Actual gravity turn (sometimes called zero lift) trajectories were calculated for the IRIS vehicle at launch angles of 70, 80, and 88 degrees, each at net payloads of 75, 100, and 200 pounds. The launch angles were chosen to show a very broad flight regime, while the payloads were estimated to be minimum, nominal and maximum.

Dispersion

Some IRIS dispersion data are contained in Reference 1, but to achieve consistency with the other vehicles in the study, calculations were conducted on the IBM 704 digital computer, using the results of the performance calculations as a starting point. The following values were used as one sigma variations at burnout:

a. Pitch flight path angle, $+2°$
b. Yaw flight path angle, $\pm 2°$
c. Velocity, ± 1 per cent

Trajectories were computed from burnout to impact for each of these conditions for an 85° launch angle and a nominal payload. Dispersion was then calculated as the root mean square of the individual contributions. Step one dispersion was not considered, since step one range to impact is small. The dispersion radius for step two is thus approximately 18 nautical miles.

The dispersion data presented here are too small if arbitrary winds at launch are considered. Since wind dispersion can be a very serious problem for an unguided vehicle of this type, detailed study would be required.

Actual and Ideal Velocity

Incremental actual and ideal velocity as a function of payload are shown in Figure 17 and Figure 18, while Figure 19 shows both this data, at nominal payload, and the velocity losses due to drag and gravity. All this information is presented for an 88° launch angle. Incremental actual velocity was obtained directly from the computer runs. Incremental ideal velocity was computed in the standard manner:

$$\Delta V_{ID} = (I_{sp})_{AVG} \, g_s \, \ln \mu$$

where average specific impulse was determined by integration of the thrust-time trace from the computer runs, and dividing the result by the consumed weight.
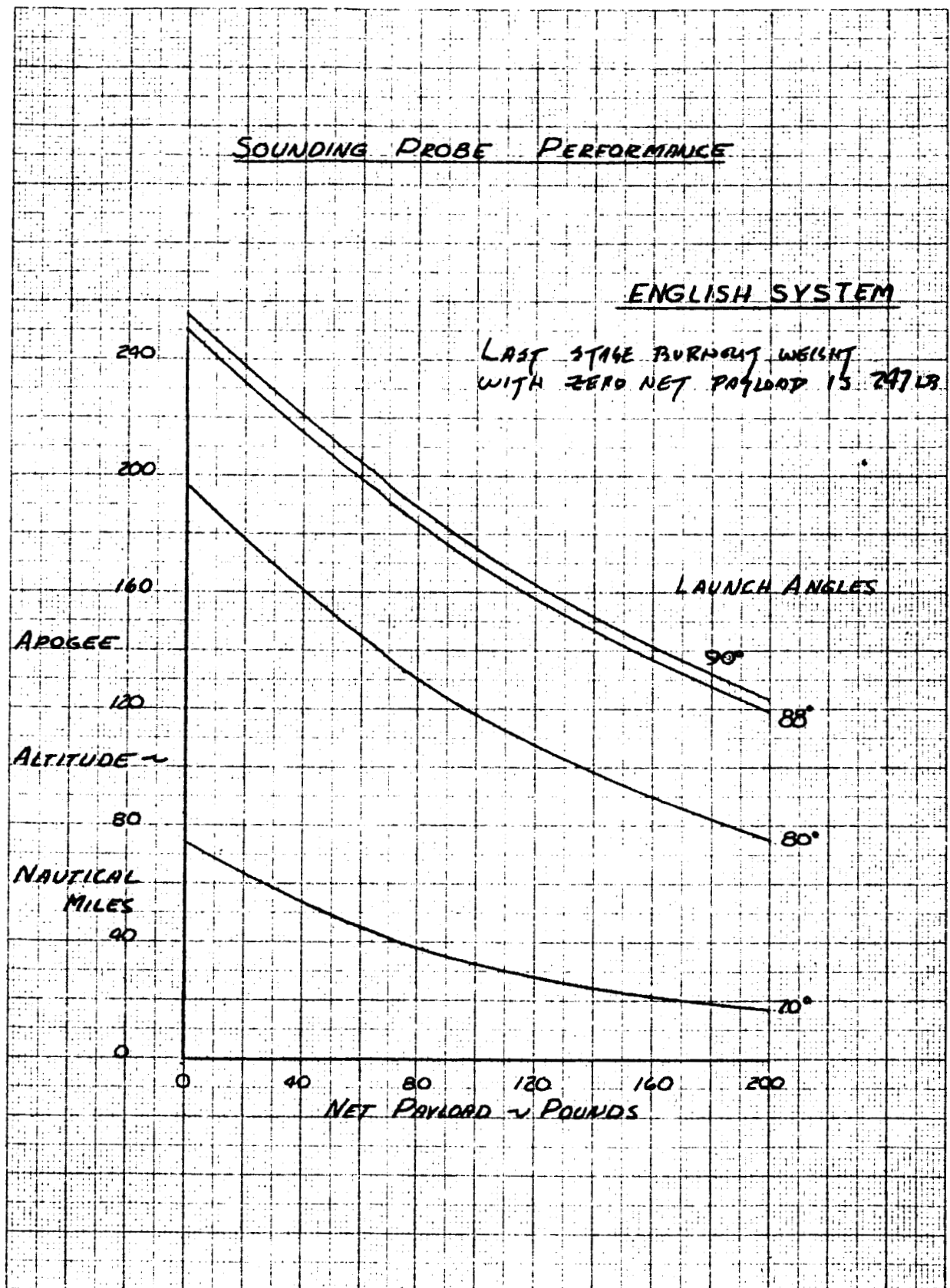
FIGURE 3 SOUNDING PROBE PERFORMANCE (ENGLISH SYSTEM)
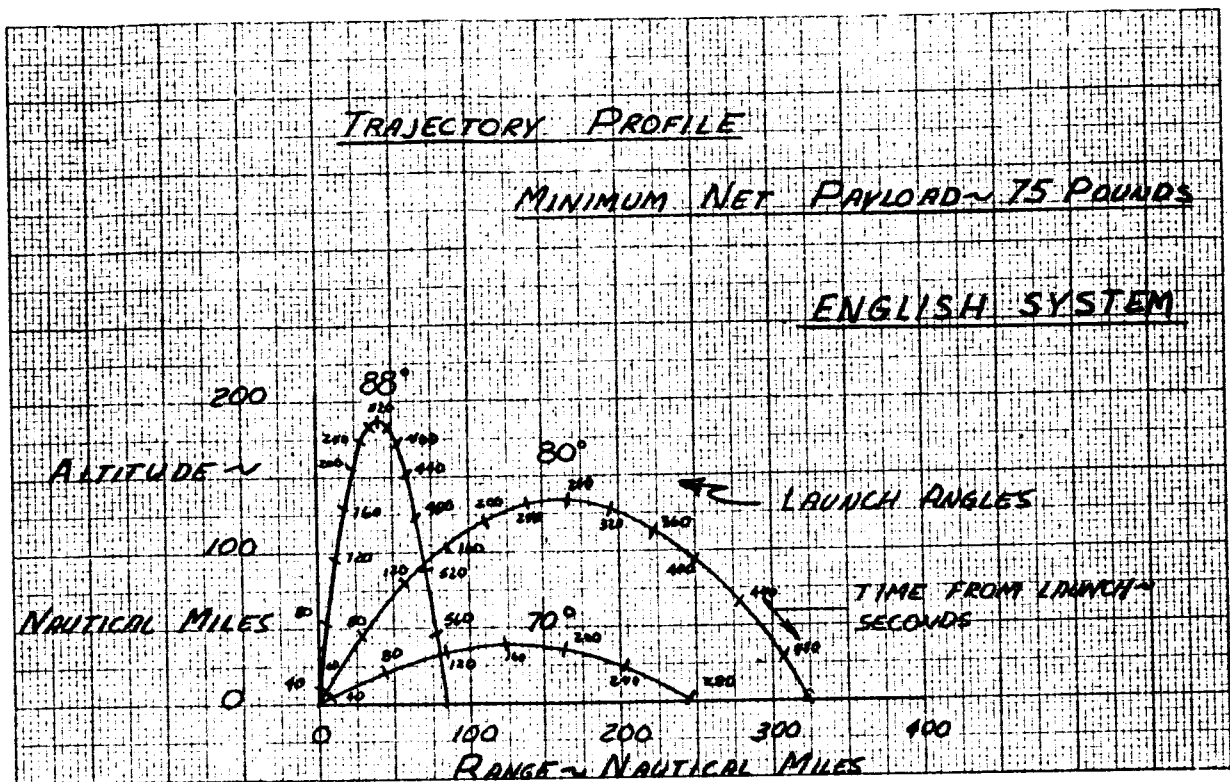
FIGURE 4   TRAJECTORY PROFILE, MINIMUM NET PAYLOAD 75
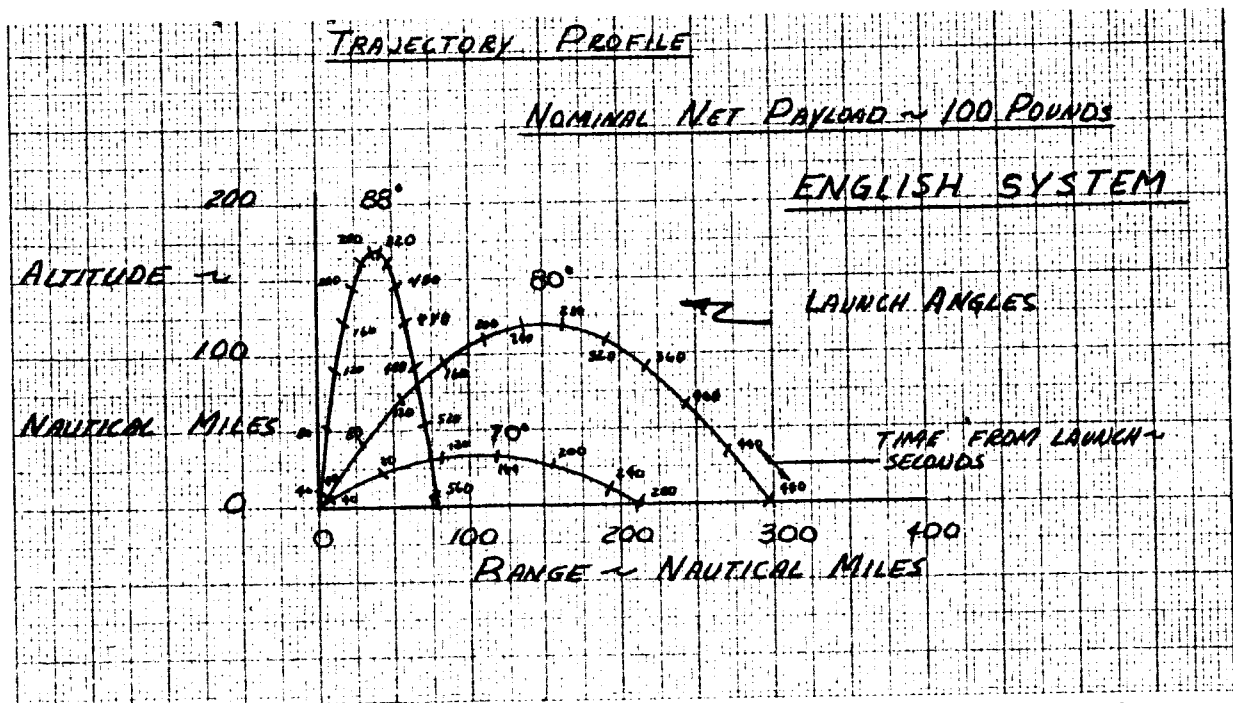POUNDS (ENGLISH SYSTEM)

FIGURE 5   TRAJECTORY PROFILE, NOMINAL NET PAYLOAD 100
POUNDS (ENGLISH SYSTEM)
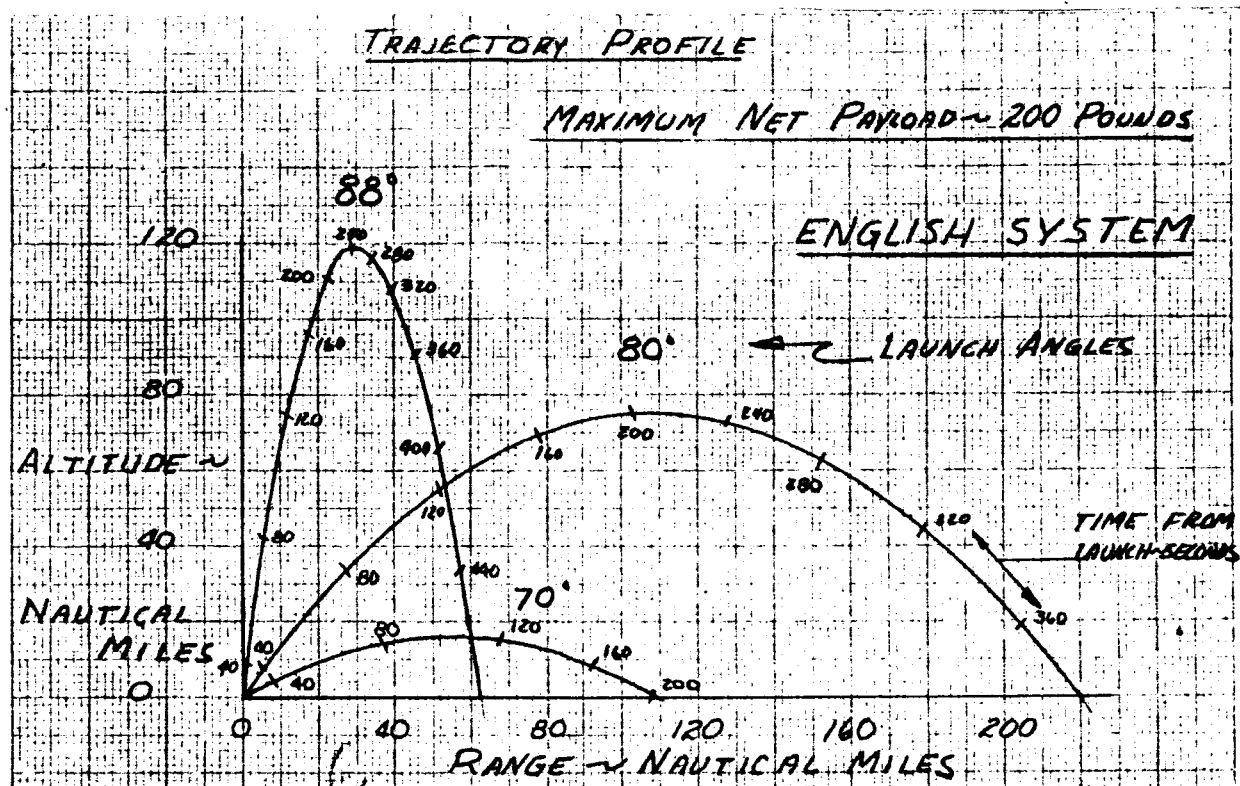
FIGURE 6   TRAJECTORY PROFILE, MAXIMUM NET PAYLOAD 200
            POUNDS (ENGLISH SYSTEM)
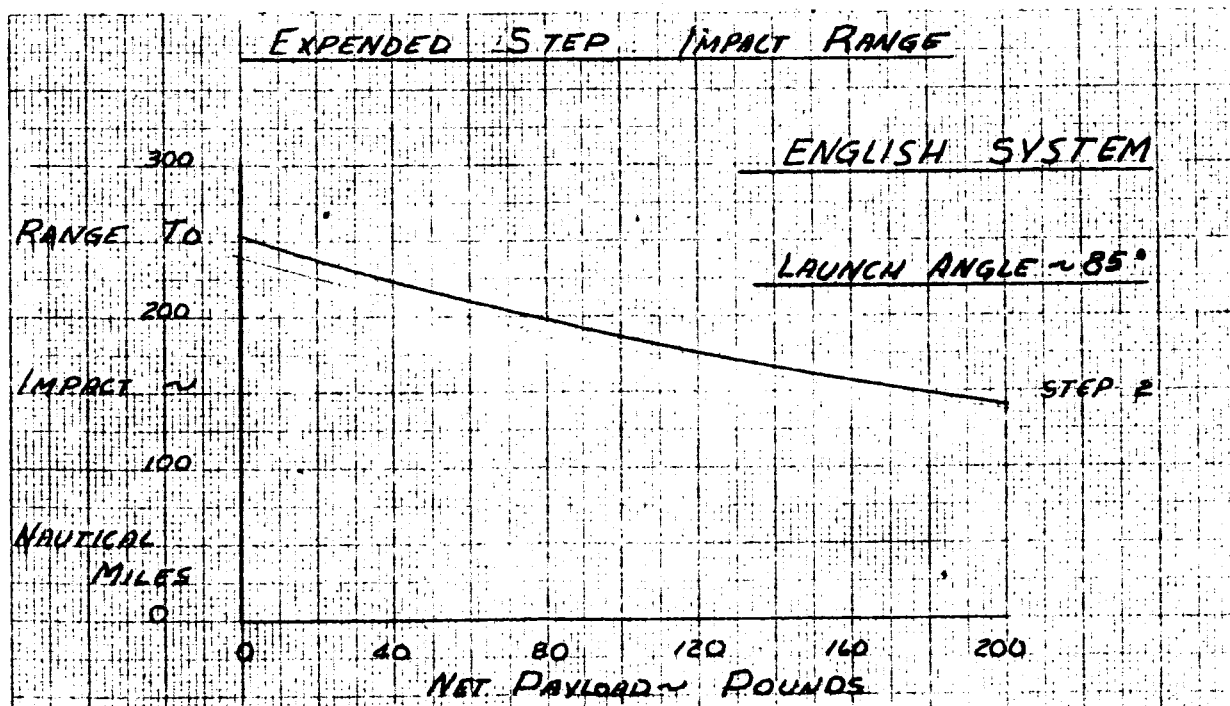
FIGURE 7   EXPENDED STEP IMPACT RANGE (ENGLISH SYSTEM)
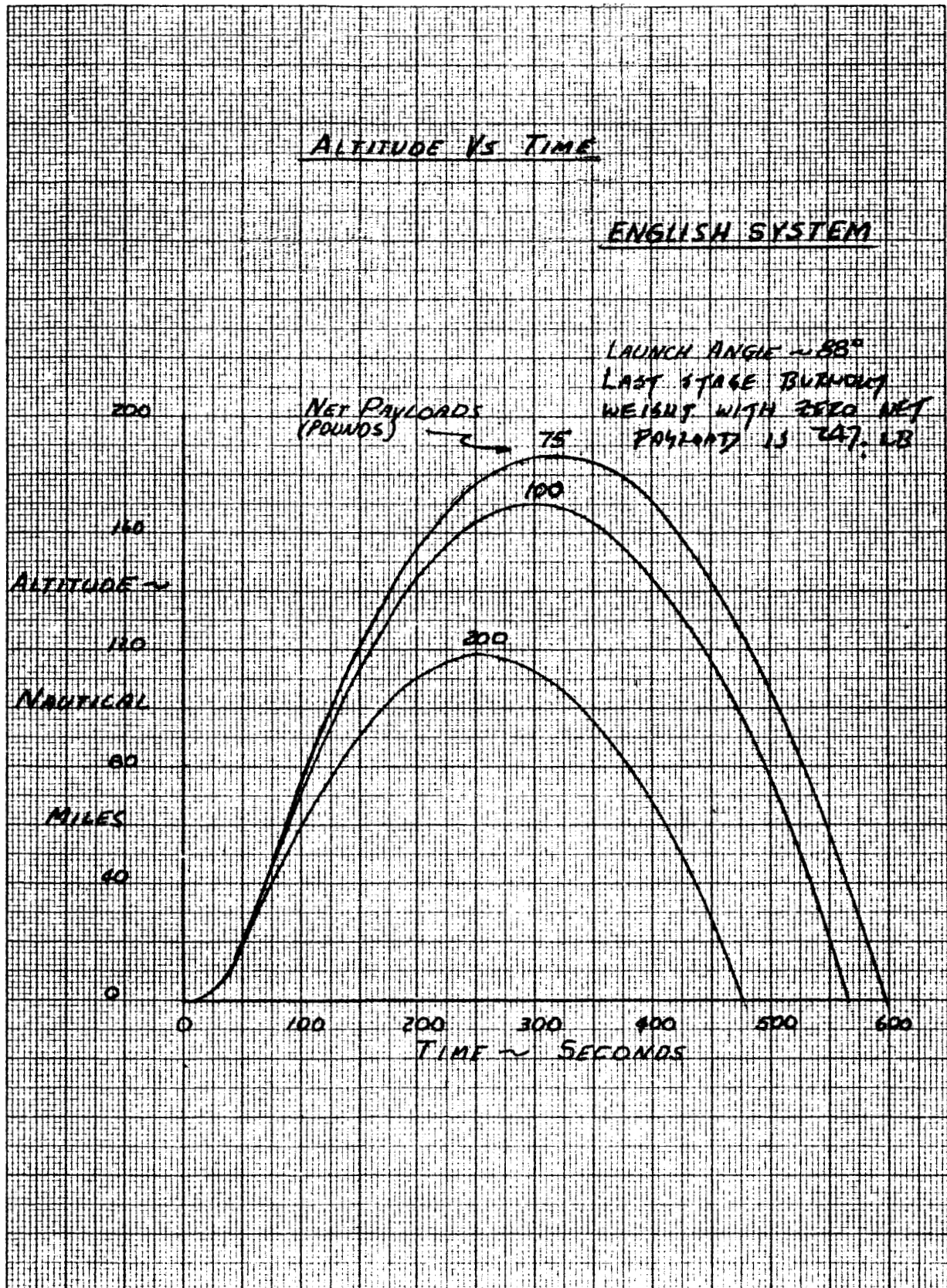
FIGURE 8   ALTITUDE VS. TIME (ENGLISH SYSTEM)
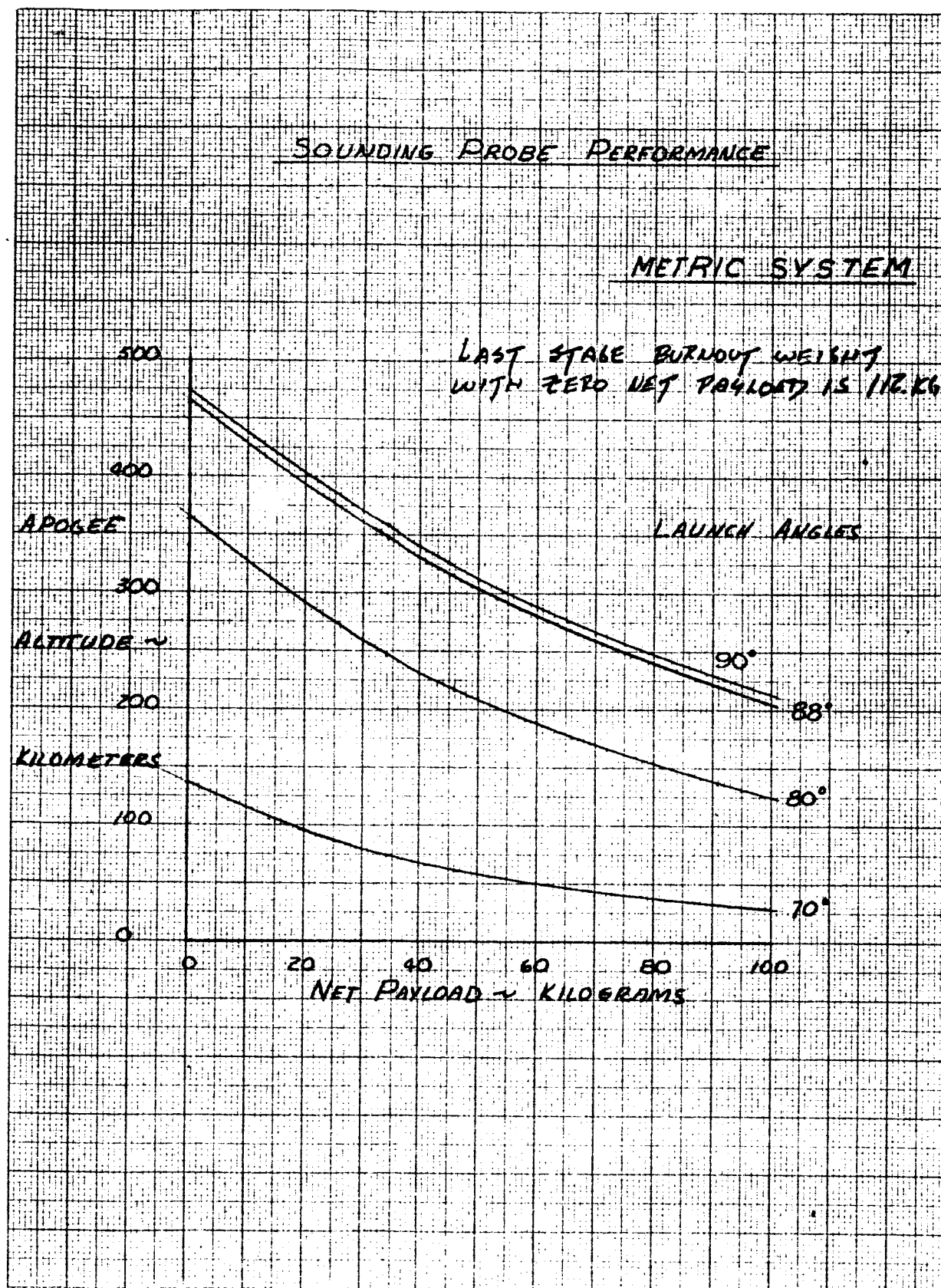
FIGURE 9   SOUNDING PROBE PERFORMANCE (METRIC SYSTEM)
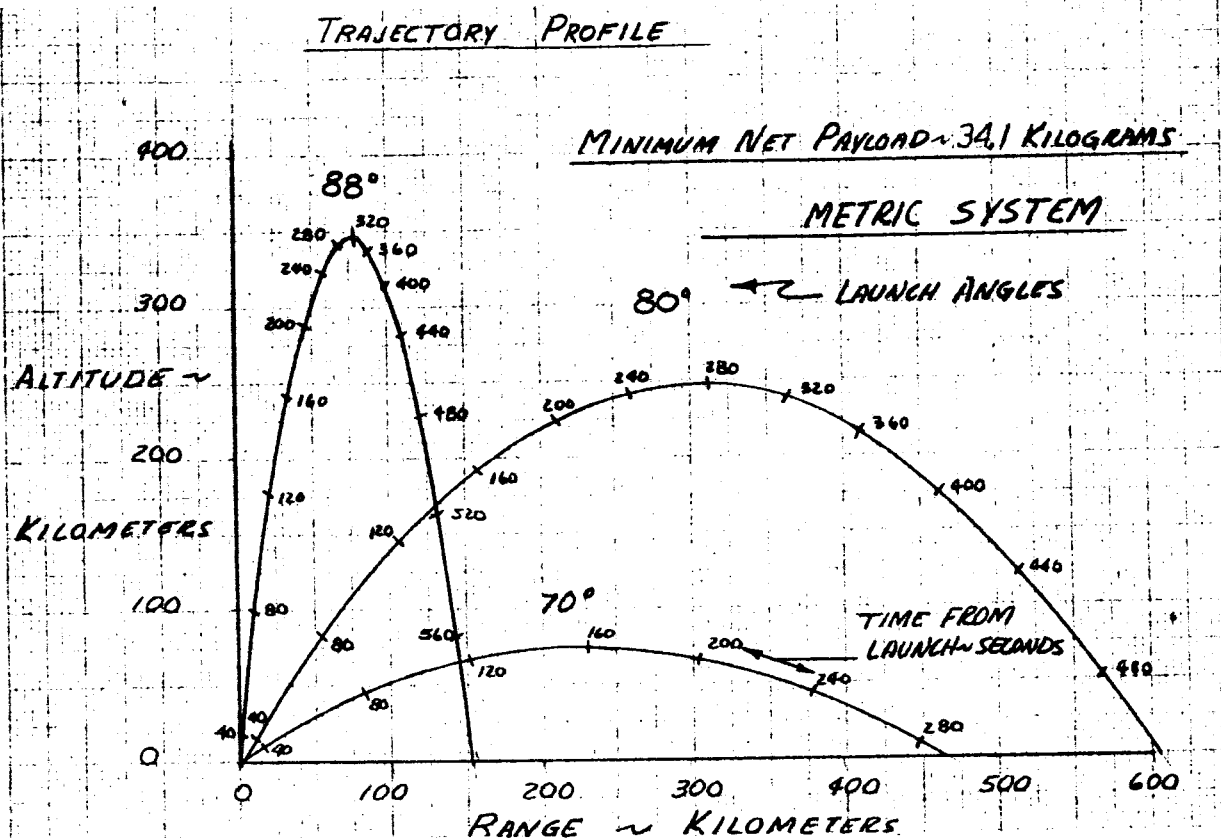
FIGURE 10   TRAJECTORY PROFILE, MINIMUM NET PAYLOAD 34.1
            KILOGRAMS (METRIC SYSTEM)
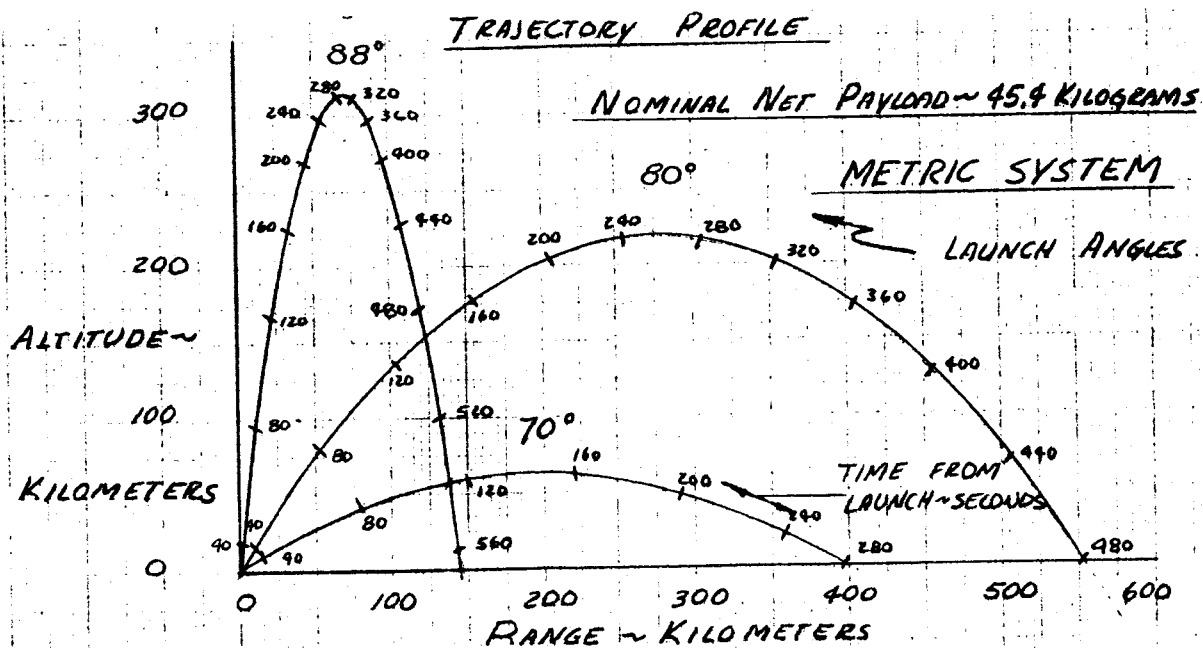
FIGURE 11   TRAJECTORY PROFILE, NOMINAL NET PAYLOAD 45.4
            KILOGRAMS (METRIC SYSTEM)
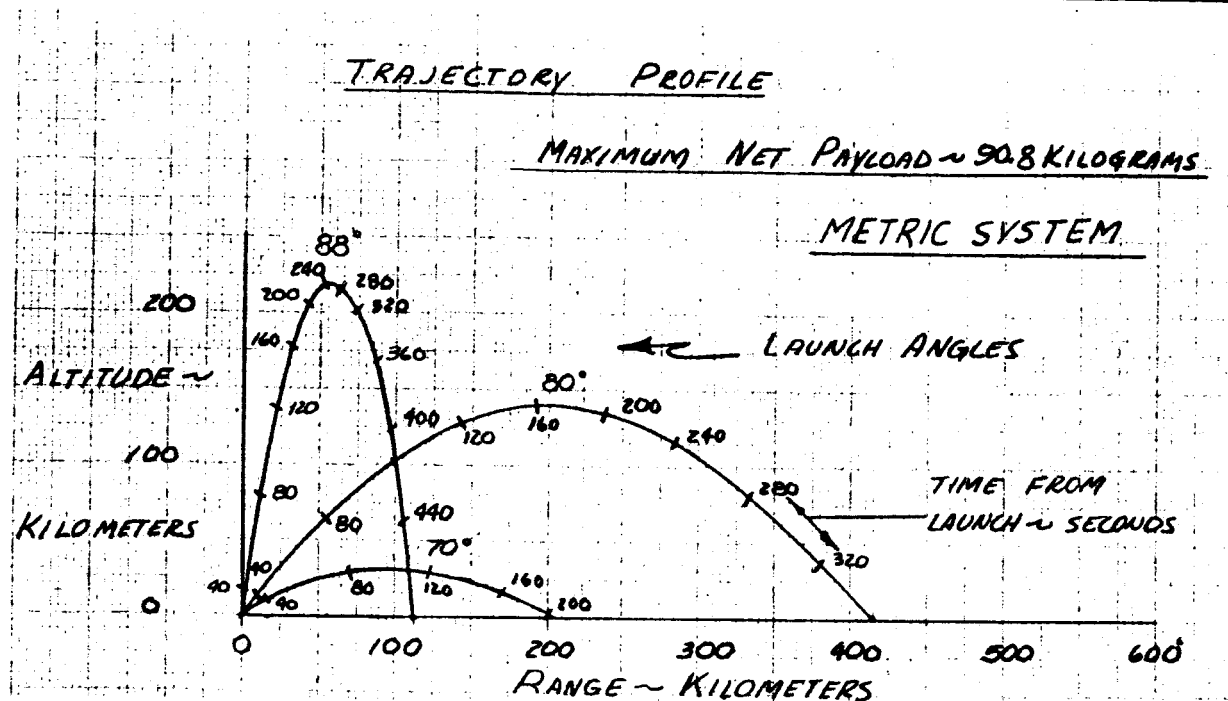
FIGURE 12   TRAJECTORY PROFILE, MAXIMUM NET PAYLOAD 90.8
KILOGRAMS (METRIC SYSTEM)
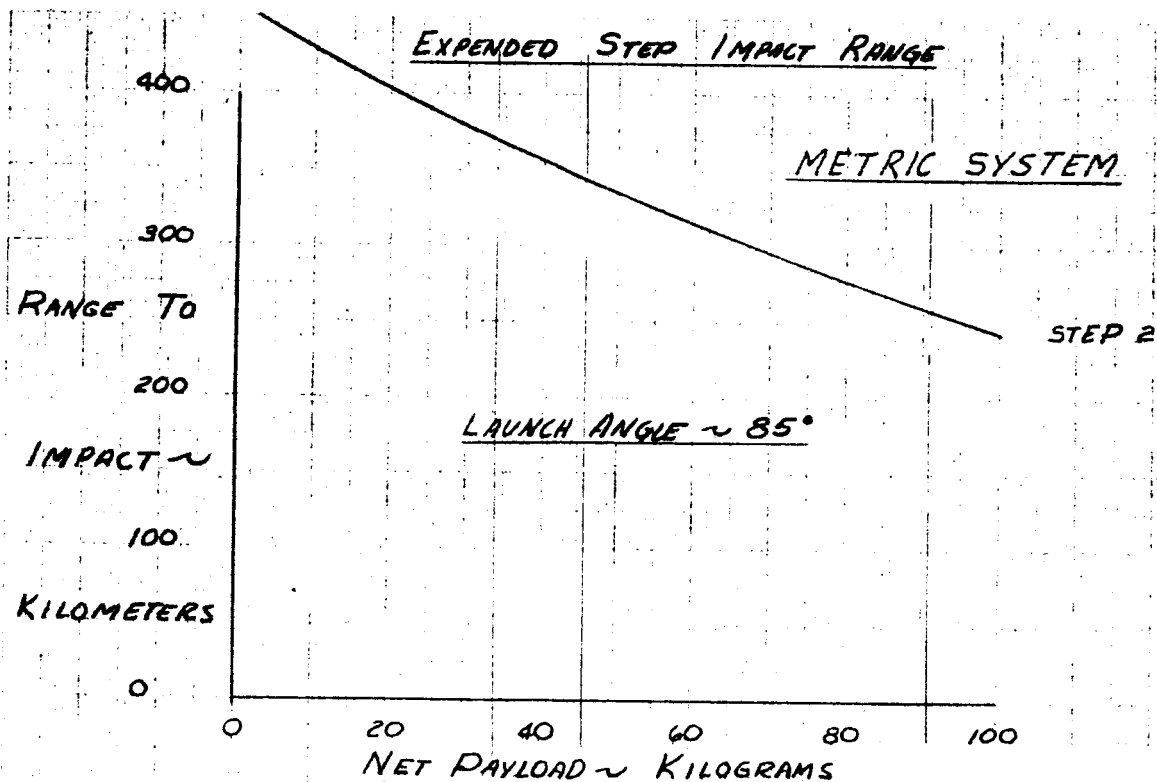
FIGURE 13   EXPENDED STEP IMPACT RANGE (METRIC SYSTEM)
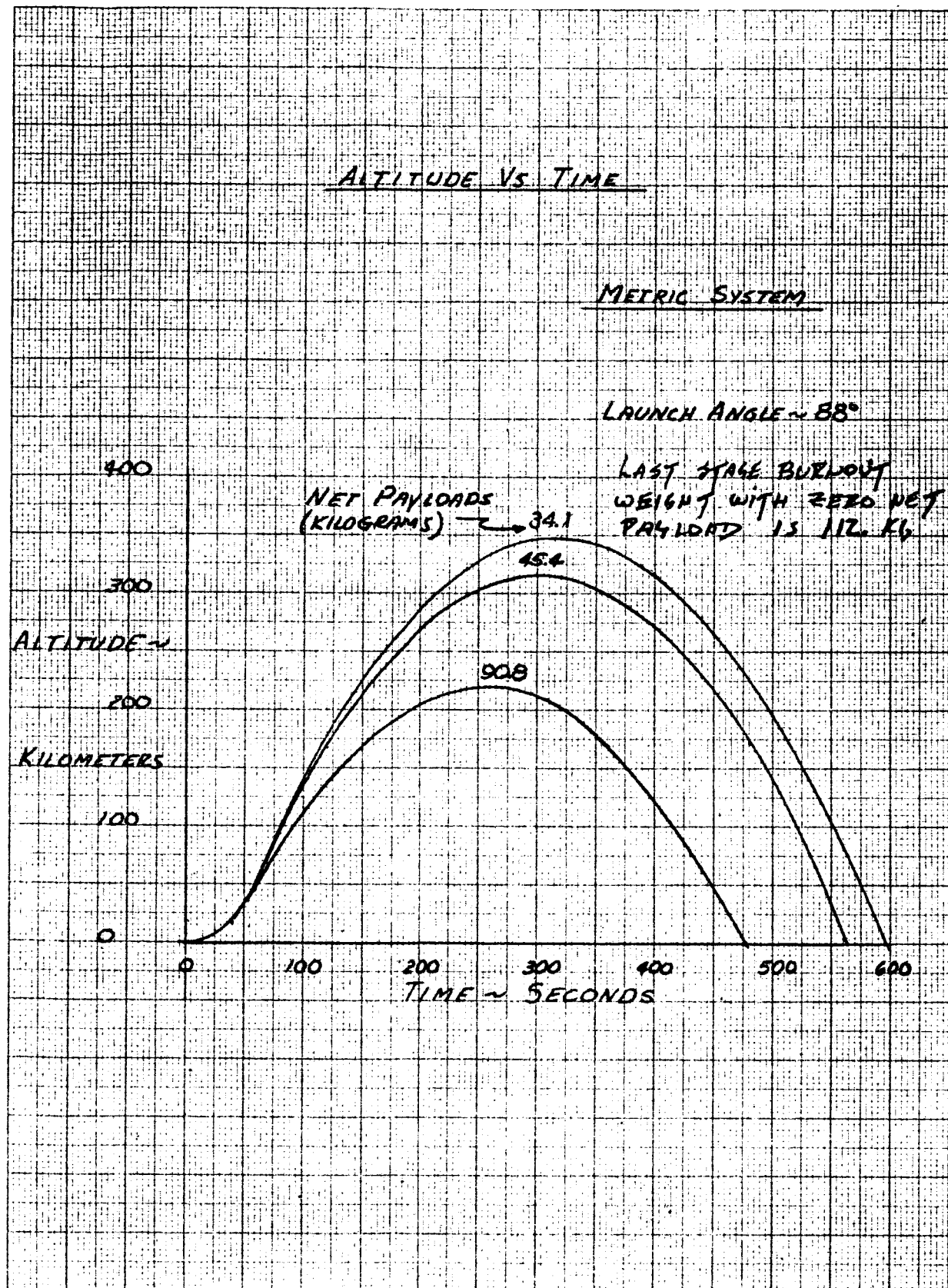
FIGURE 14   ALTITUDE VS. TIME (METRIC SYSTEM)

TIME ABOVE 100 NAUTICAL MILES

100 NAUTICAL MILES = 185.2 KILOMETERS

LAST STAGE BURNOUT WEIGHT WITH ZERO NET PAYLOAD IS 247 LB (112 KG)

LAUNCH ANGLES

88°

85°

80°

TIME ~ SECONDS

NET PAYLOAD ~ POUNDS

NET PAYLOAD ~ KILOGRAMS
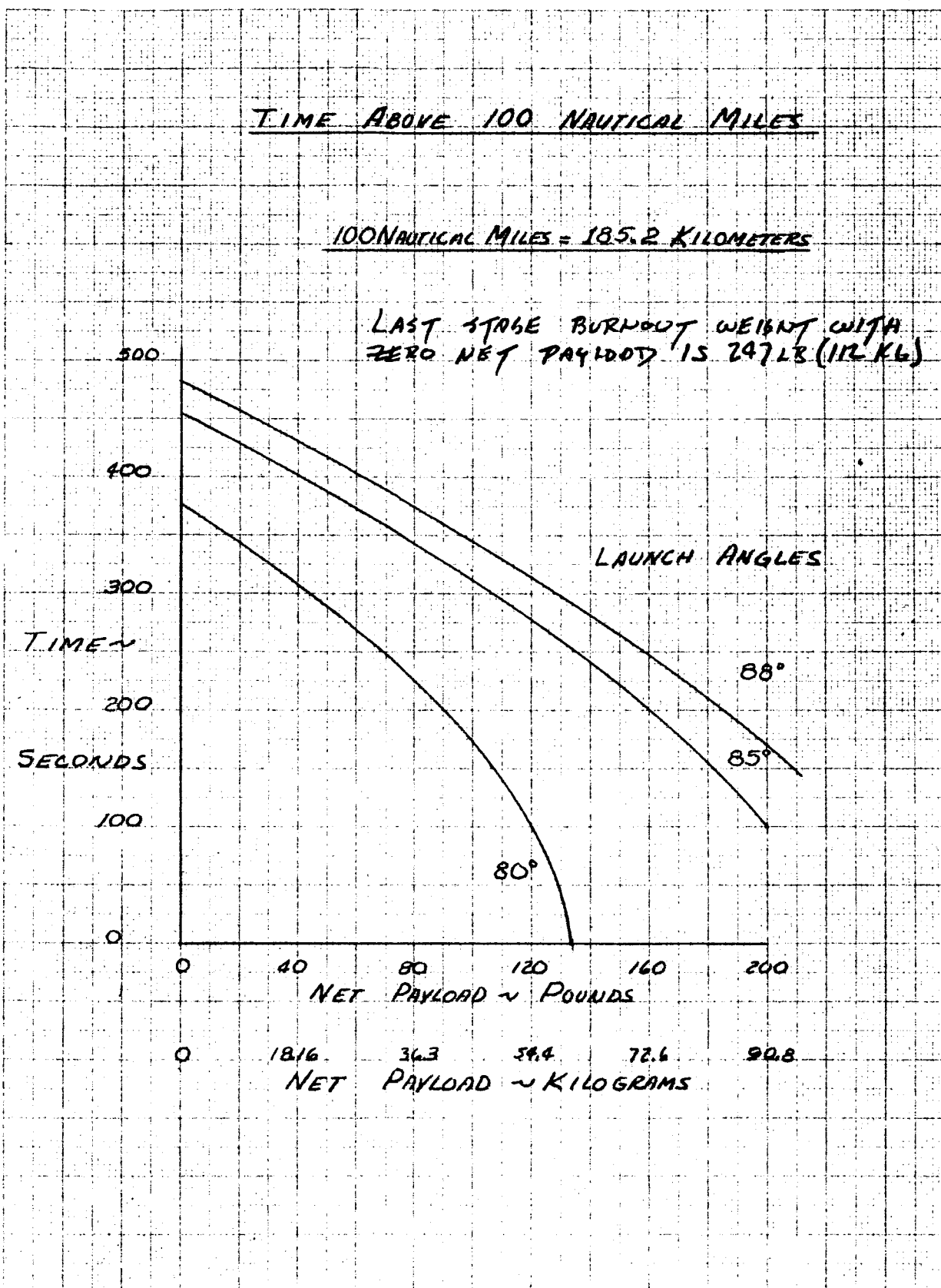
FIGURE 15   TIME ABOVE 100 NAUTICAL MILES
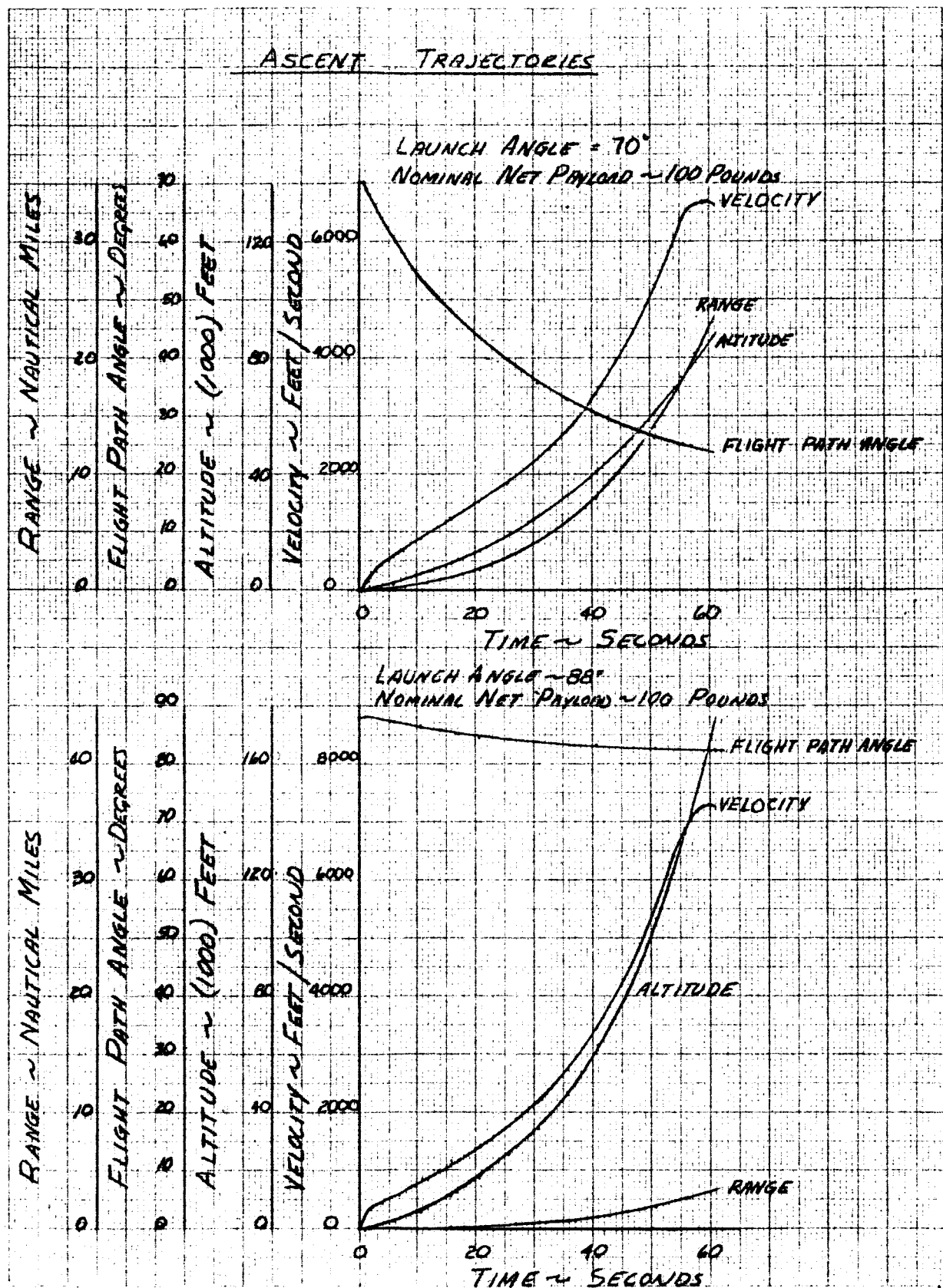
FIGURE 16   ASCENT TRAJECTORIES (70 AND 88 LAUNCH)
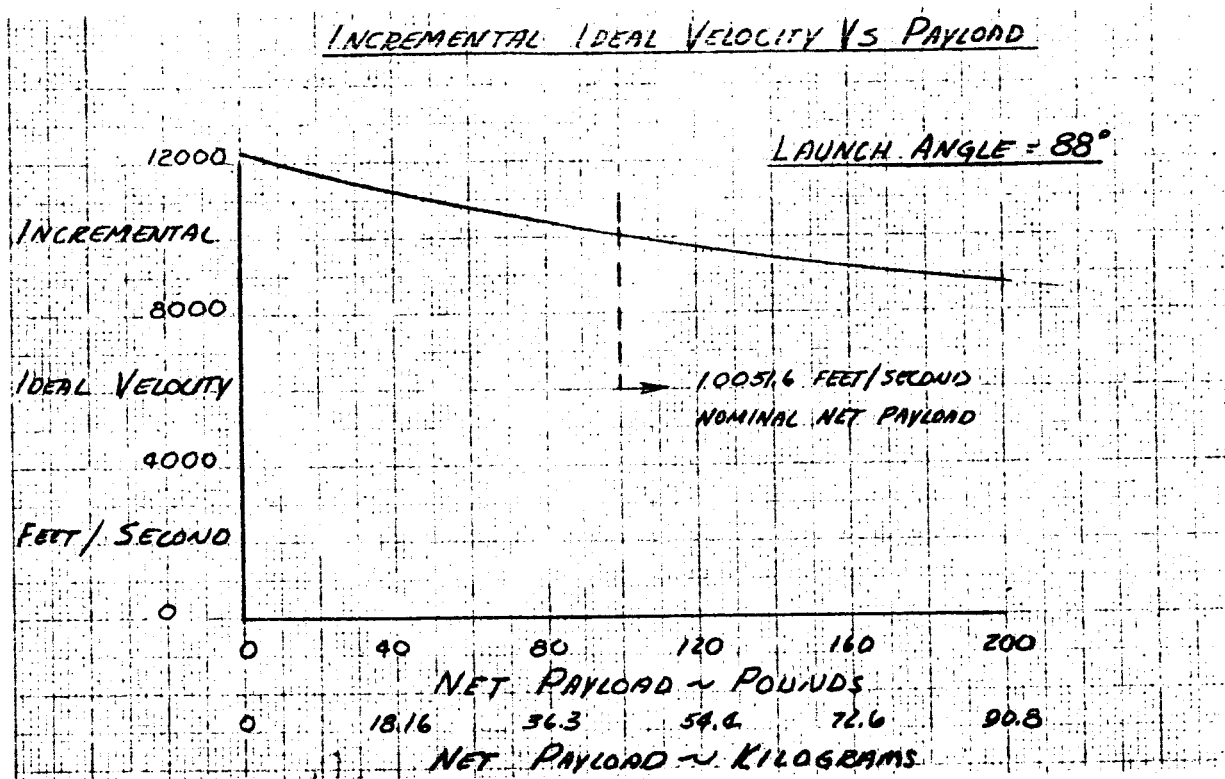
FIGURE 17 INCREMENTAL IDEAL VELOCITY VS. PAYLOAD
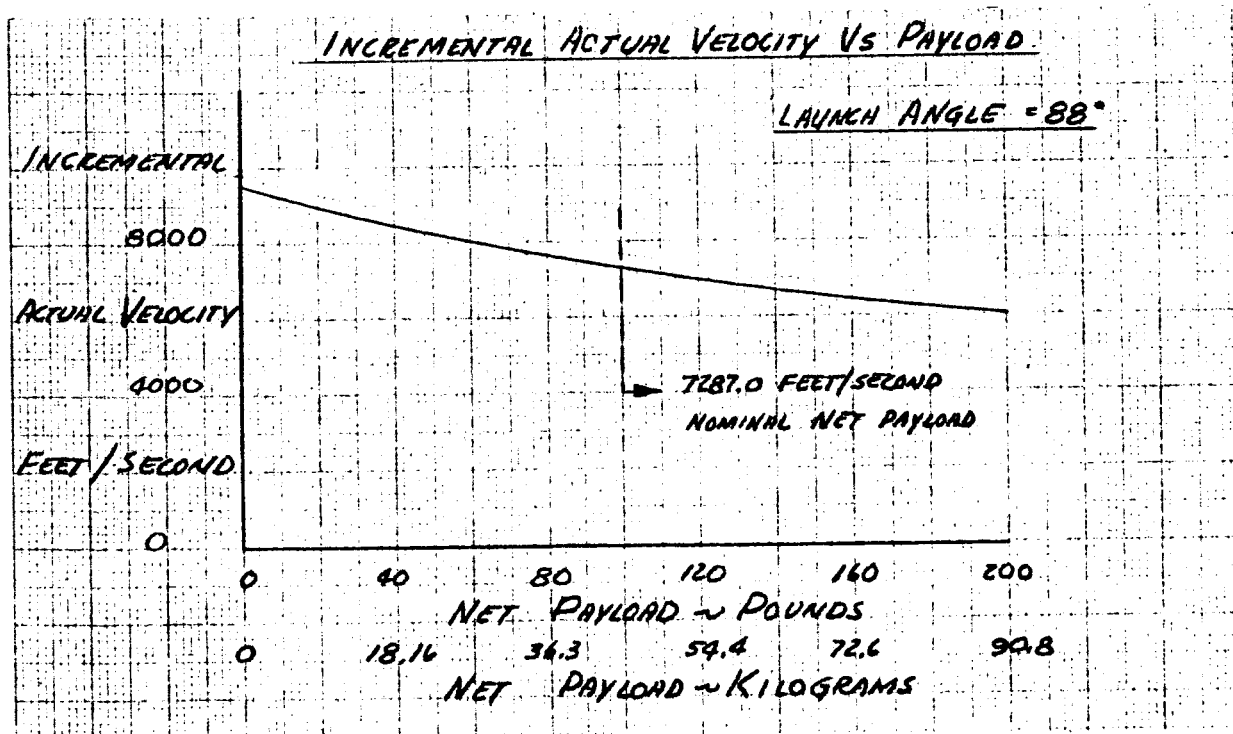
FIGURE 18 INCREMENTAL ACTUAL VELOCITY VS. PAYLOAD

IDEAL AND ACTUAL
VELOCITY TABULATION

LAUNCH ANGLE = 88 °		NOMINAL NET PAYLOAD = 100 POUNDS		
PHASE OF FLIGHT	IDEAL VELOCITY INCREMENT (FT/SEC)	VELOCITY LOST TO DRAG AND GRAVITY (FT/SEC)	COAST VELOCITY LOST (FT/SEC)	ACTUAL VELOCITY INCREMENT (FT/SEC)
STAGE 1 BOOST*	353.	26.0	0	327.
STAGE 2 BOOST	9699	2739.	0	6960
TOTALS	10052.	2765.	0	7287.

*Both first and second step motors ignite at launch. Stage one boost
ends at burnout of the step one motor.

FIGURE 19. IDEAL AND ACTUAL VELOCITY TABULATION

BASIC DATA

Weight

Detail Weight Breakdown

   The following detail weight breakdown was used as a basis for all weight, c.g., and inertia calculations. These data were taken from an Atlantic Research letter to K. M. Russ (Chance Vought Corporation) dated 20 January 1961 with the following modifications:

   1. Nose cone weight rounded to 12 pounds to agree with NASA data.

   2. Consumed weight of 967 pounds used to agree with NASA data.

   A consolidation of this data into major components for c.g. and local moment of inertia presentation is tabulated on the next page, while over-all vehicle weight, c.g., and inertia data versus time are presented in Figures 20 and 21.

IRIS Detailed Weight Breakdown

Second Step

 Inert Weight

Nose Cone	12.0
Head Closure	12.5
Insulation	30.0 (60 lbs. total - assume 30 lbs. consumed)
Motor	116.0
Nozzle	35.5
Fin Assembly	41.0
Total Inert	247.0

Consumed Weight:

Propellant & Inhibitor	937.0
Insulation	30.0
Total Loaded Weight	1214.0

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First Step

(For information only - not used in c.g. or inertia calculations)

Inert Weight:

Inert Motor (7)	91.0
Structure	40.5
Total Inert	131.5
Consumed Weight:	73.5
Total Loaded Weight	205.0
Launch Gross (less payload)	1419.0

Center of Gravity and Inertia Data

Item	Weight Pounds	C.G.,In. from Ref. Sta. 0.0	Local Roll Moment of Inertia Slug-Ft.^{2}	Local Pitch or Yaw Moment of Inertia Slug-Ft.^{2}
Second Step				
Nose Cone	12.0	46.0	.0995	1.430
Fins	41.0	224.8	2.181	2.047
Motor Case	194.0	168.0	1.508	89.02
Step Total-Empty	247.0	171.5		
Consumed Weight	967.0	153.9	3.759	383.1
Step Total-Loaded	1214.0	157.5		
Payloads:				
Minimum	75.0	56.0	.3077	5.681
Nominal	100.0	56.0	.4102	7.574
Maximum	200.0	56.0	.8204	15.15

Notes:

(1) Local roll moment of inertia of fins is about vehicle centerline.
(2) Motor c.g. data from Reference 2.
(3) Station 0.0 is tip of nose cone of 278.7" long vehicle as shown in Reference 2.
(4) All other data calculated.

Aerodynamics

The basic aerodynamic data at zero angle of attack are shown in Figure 22. For payloads of 75 pounds, the vehicle borders on instability at burnout and this payload should be considered the minimum. For relatively small angle of attack, some forward shift of the center of pressure will probably occur but the changes in stability will be slight. The rocket will probably be unstable at very large angles of attack. Drag will increase rapidly with angle of attack for this rocket.

Propulsion

Since the rocket motor ballistic data for some of the motors used on the 18 vehicles of this sounding rocket study series are classified, all of the ballistic data, both classified and unclassified, were consolidated into report no. AST/E1R-13336 so that all of the individual vehicle reports would remain unclassified. The classification of the Iris motor is Confidential.
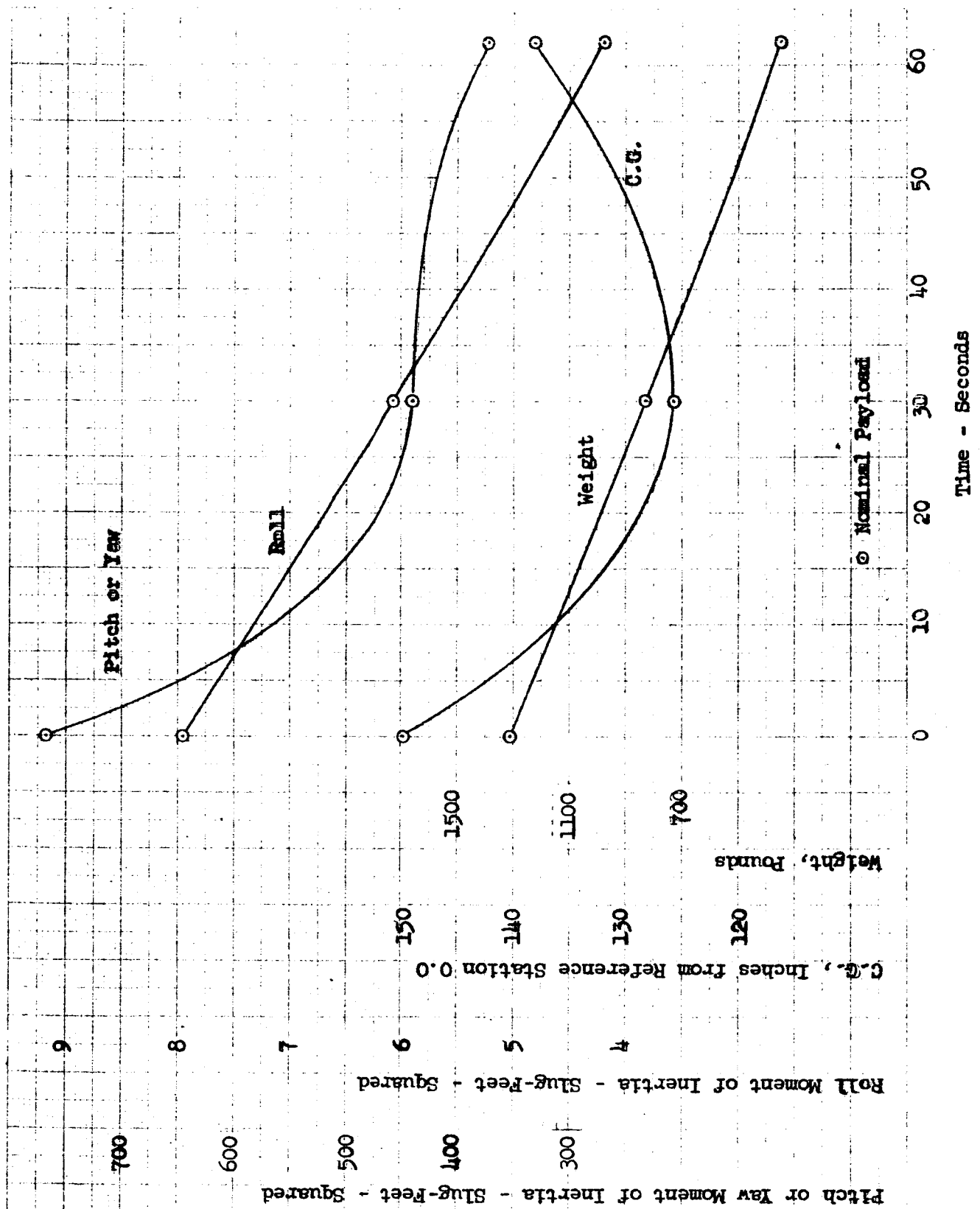
FIGURE 20   WEIGHTS, CENTER OF GRAVITY AND MOMENTS OF
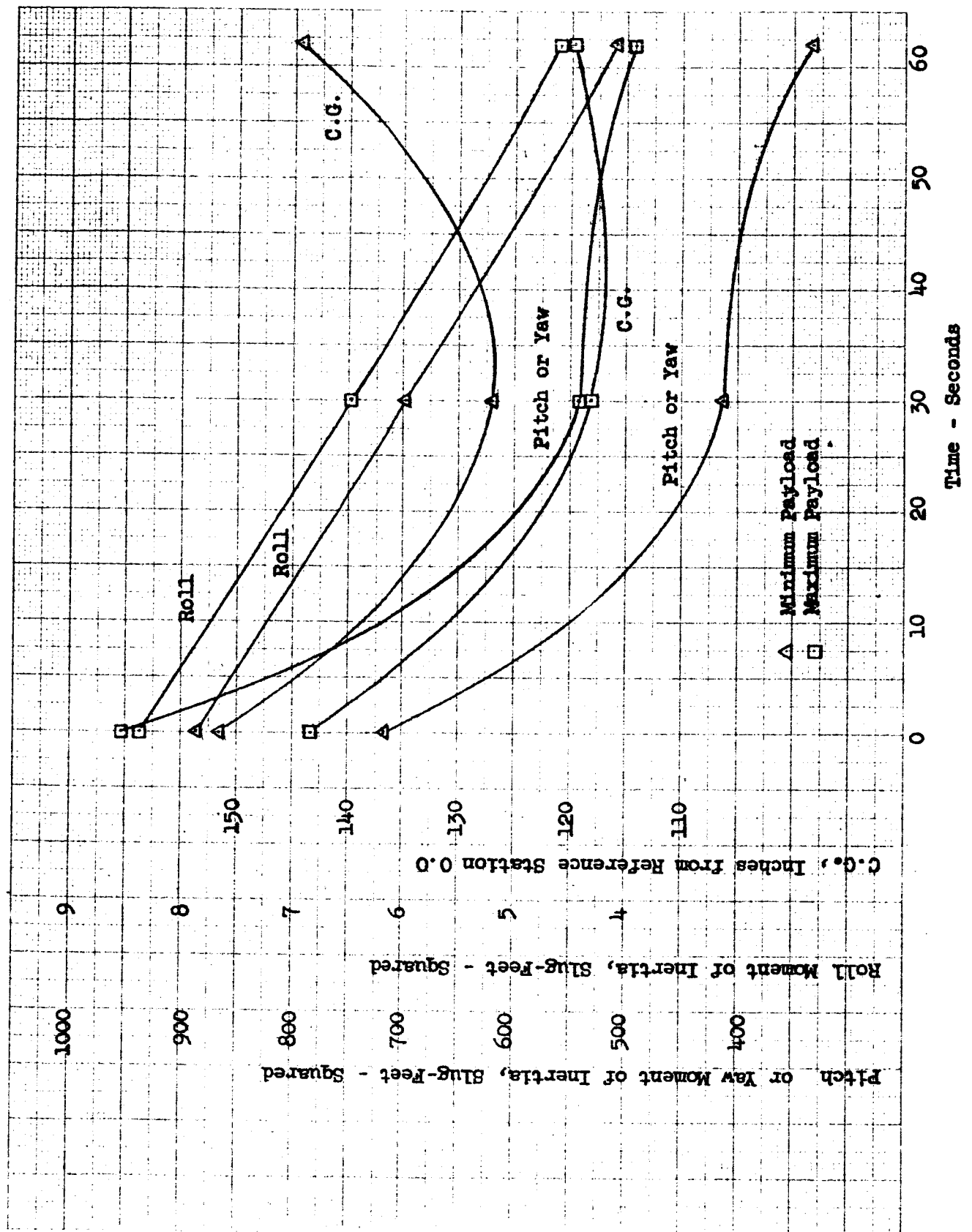INERTIA VS. TIME (STAGE 2)

FIGURE 21   CENTER OF GRAVITY AND MOMENTS OF INERTIA
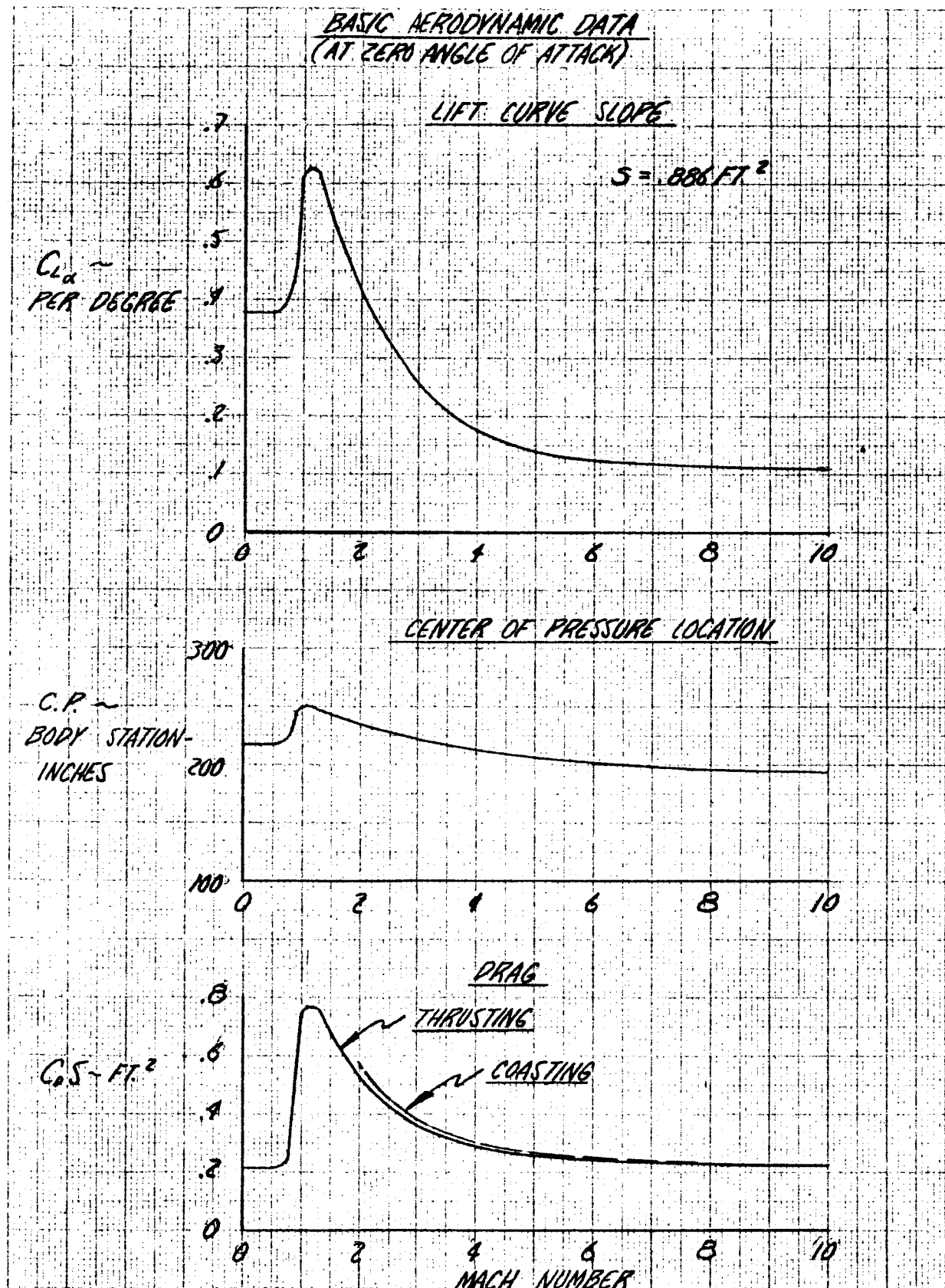VS. TIME (STAGE 2)

FIGURE 22   BASIC AERODYNAMIC DATA (LIFT, CP, DRAG)

ENVIRONMENT

Axial Acceleration

The axial acceleration time history for the IRIS vehicle at
a launch angle of 88° is shown in Figure 23.

Roll Rate

The IRIS roll rate time history is shown in Figure 24. Tangential
and centripetal accelerations acting on the payload in this case will proba-
bly be a minor consideration. The roll rate shown can indirectly affect
the loads through pitch-roll coupling.

Structural Dynamic Analysis

The flexible vehicle structure is subjected to a number of
loading environments which produce significant dynamic responses.
The load inputs occur from ground handling, launch, atmospheric dis-
turbances, stage separation, and structural and thrust misalignments.
The spin stabilized vehicle experiences additional loading phenomena
arising from dynamic coupling. Atmospheric disturbances in the form
of winds and gusts require an extensive analysis for structural loads
determination. This involves a trajectory analysis of the flexible vehicle,
taking into consideration time variations in weight and aerodynamic load
distributions. Atmospheric winds and gusts are defined statistically
so that ultimately the analysis produces a missile loading criteria in
terms of probability of structural failure.

The weight distribution may be obtained from available infor-
mation but other information necessary to determine the structural
dynamic characteristics of the IRIS vehicle has not been received.

Vibration

In order to obtain the vibration environment in the payload
compartment, it is necessary to know the vibration characteristics of
the sources, such as rocket motor(s), aerodynamic boundary layer
noise, and launch noise. The structure-borne and airborne transmis-
sion path characteristics of these sources of vibrations must also be
known in order to establish payload base vibration environment. Pay-
load base input vibrations normally would be expressed as a function of
vibration amplitude versus frequency for significant flight times, and

with the characteristics of the vibration (i.e., sinusoidal, random, mixed) indicated. The resulting payload vibration environment will depend upon the structural dynamic response characteristics of the payload input vibration environment.

No payload compartment vibration data had been received at the time of writing this report. Vibration data collected on Iris NASA 5.03 will be available from the Goddard Space Flight Center in the near future.

Temperature

External Temperature

To determine the temperature effects on this vehicle, it was necessary to select a given trajectory and specific components to be investigated. The 70° launch angle and nominal payload were selected as a limiting condition which would emphasize possible mission restrictions that result from skin heating. A vehicle which has been used satisfactorally at launch angles above 80° might be inadequate for a 70° launch. This is shown to be the case for the Iris vehicle from the temperature curves of Figure 25.

The components investigated include the nose cone and fin stagnation areas, the nose cone fairing in the payload area, and the fin panels as shown in Figure 25. While these areas normally experience maximum heating, this does not imply that other areas on the vehicle, such as rocket cases, do not require investigation for a particular mission.

Skin gages shown in Figure 25 were obtained from the manufacturer. For the nose tip and fin leading edges which contained internal heat sinks locally, average stagnation temperatures were obtained by treating the tip as a one dimensional problem where the skin thickness was arbitrarily computed as one quarter of the length of the heat sink. This approach was considered satisfactory but the resulting temperatures in this area are recognized as being approximate.

The physical properties for the transient temperature analysis, using digital computer methods, are shown below:

Material	Aluminum	Steel	Magnesium
Density (lb/ft^3):	173.	489.	106.
Specific Heat (Btu/Lb- °F):	.24	.21	.28
Emissivity:	.35	.8	.6

To properly evaluate the structural reliability of any vehicle, the load-temperature relationship with respect to time must be considered. These relationships cannot be adequately defined until a specific mission requirement has been selected. For example, a component may experience severe reduction in strength allowables due to temperature, but if the elevated temperature occurs at times when the load is negligible, the condition may be acceptable.

Temperature curves (B) and (D) of Figure 25 show that the aluminum nose cone and the magnesium fin panels are not adequate for this particular mission (70° launch). Use of higher launch angles would reduce these temperatures, or material substitutions could be made which would be acceptable for the 70° mission. For example, a fiberglass or steel nose cone could replace the present aluminum component. The existing structure, however, is adequate for near vertical launches for which the skin temperatures will be significantly lower. The wrapped steel fin leading edge is satisfactory for the mission shown.

Internal Heating of Payload Compartment

The payload compartment temperature, while the vehicle is on the launch pad, is a function of the ambient temperature, location of the launch pad, time on the launch pad, and the heat output of the payload.

To determine the payload compartment temperature on the launch pad, an average payload of one hundred (100) pounds with an area-weight ratio of 0.1 ft^2/lb. was considered. The compartment walls were assumed to be gold-coated (due to the low emissivity of gold) and the compartment subjected to an ambient temperature of 100°F. The compartment temperatures were calculated for payload power outputs of 10, 100, and 200 watts which correspond to payload power densities of 0.1, 1.0 and 2.0 watts/lb., respectively. The compartment temperatures were calculated considering convection, radiation, and storage of heat by the payload. Considering these conditions, payloads with a power density of 2.0 watts/lb. or above, will require additional cooling to hold the compartment temperature to 150°F or below if they remain on the launch pad with power on from one to two hours prior to launch (which is generally

not normal procedure). The usual pre-launch "power on" condition is of relatively short duration and therefore pre-launch temperature is not normally a problem. The maximum compartment temperature limit for most electronic equipment is 150°F. Additional cooling of the payload compartment, if necessary, may be accomplished by forced ventilation, cooling to a subcooled state prior to launch, and by the addition of heat sinks to the payload.

The heating of the payload compartment after launch is a function of the compartment temperature prior to launch, vehicle flight path, duration of flight, heat output of the payload, and compartment configuration.

To determine the payload compartment temperature after launch, payloads of the same magnitude as above were considered. A nominal atmospheric trajectory was used to determine the effects of aerodynamic heating on the compartment. Since the flight time of the Iris vehicle is of short duration (approximately 9 minutes), the payload compartment temperature rise due to aerodynamic heating will be small. Therefore, if the payload compartment temperature is 125°F or below, no additional cooling of the payload should be necessary. However, since the payload compartment temperature is a function of the conditions previously mentioned, the environment of each payload should be further analyzed with respect to the conditions stated in paragraph 3 prior to establishing the payload cooling requirements.
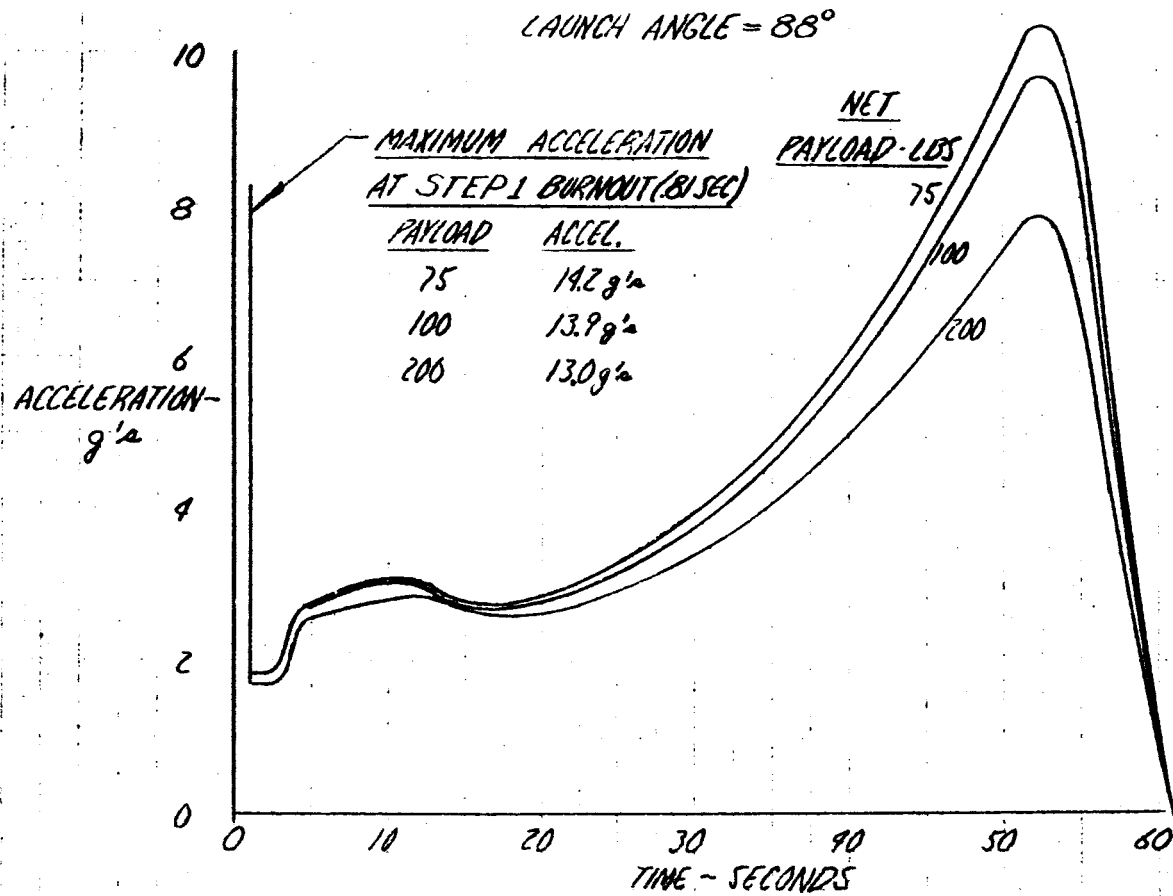
AXIAL ACCELERATION VS. TIME

LAUNCH ANGLE = 88°

FIGURE 23   AXIAL ACCELERATION VS. TIME

FIGURE 24   ROLL RATE VS. TIME

IRIS

TEMPERATURE VS. TIME

LAUNCH ANGLE = 70°
PAYLOAD = 100 LBS.
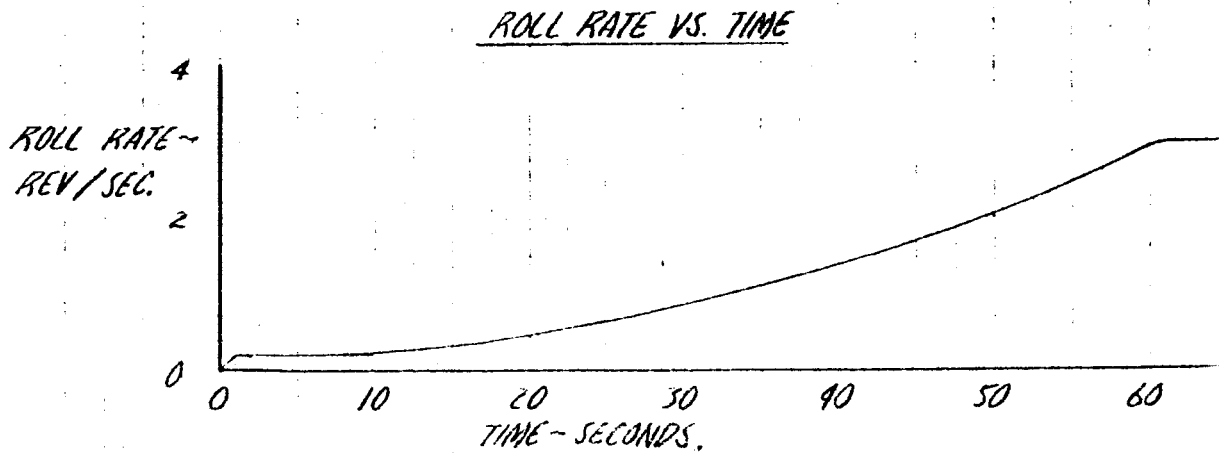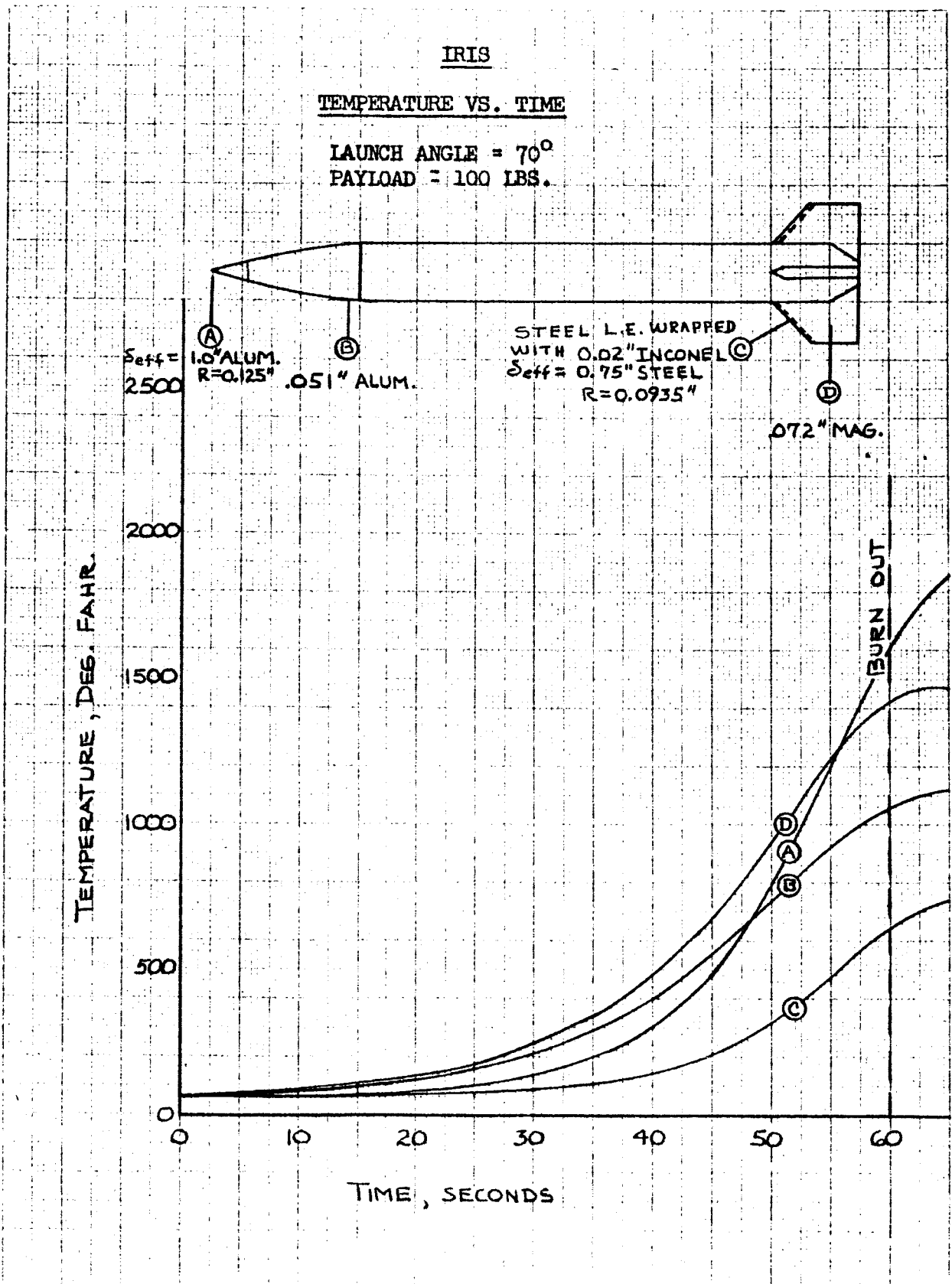
FIGURE 25   TEMPERATURE VS. TIME

OPERATIONAL FACTORS

Ground Support Equipment

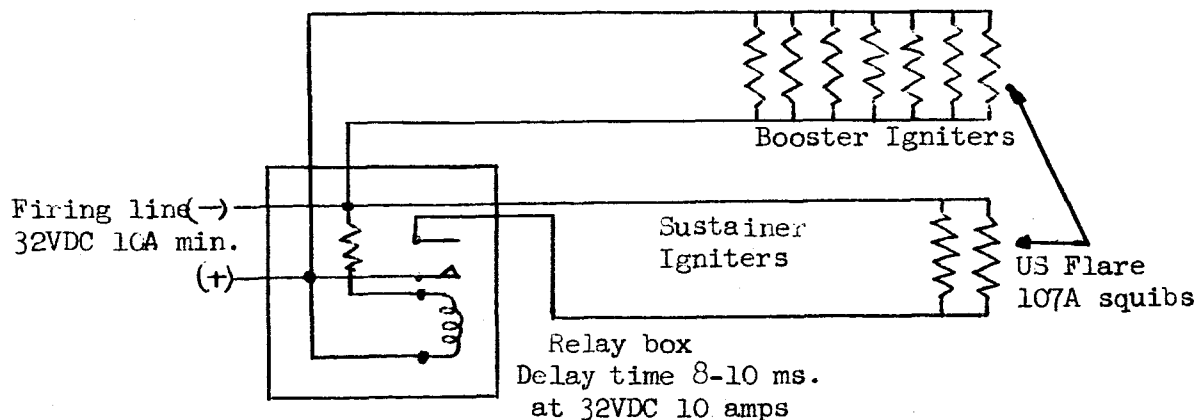Mechanical Ground Support Equipment

Launcher

It was stated that both 3-finned and 4-finned models of IRIS can be made available permitting launch from several existing 3 and 4-rail launch towers. Only the 4-finned models, however, have been assembled by Atlantic Research Corporation, which restricts launching activities to the NASA Aerobee Tower at Wallops Island, Virginia as shown in Figure 26. Details of the Wallops Island launch facility are not available. It is understood that the shops and assembly attached to the tower base building are separated by fire doors and contain a Gishold balancing machine, helium storage and pressurization equipment, etc.

The tower launches 4-finned IRIS and 4-finned Aerobees (Model 150A and Model 300A).

Electrical Support Equipment

A suitable safety circuit in the blockhouse is required for the ignition circuit as illustrated below. Two items of equipment are necessary:

1. An accurate counter for checking the time delay relay operation before assembly.

2. An igniter tester that will allow less than 0.20 amperes to flow through the squib.

Booster Igniters

Firing line (-)
32VDC 10A min.
(+)

Sustainer
Igniters

US Flare
107A squibs

Relay box
Delay time 8-10 ms.
at 32VDC 10 amps

Instrumentation

      Performance instrumentation data has been accumulated
from three IRIS flights. This data includes combustion chamber pres-
sures, accelerations, roll rates, skin temperatures, partial vehicle
aspect (from magnetometers and solar cells), vibration environment, and
pitch and yaw (angle of attack components). Additional instrumentation
may be desired on future flights to supplement payload data, verify
trajectory characteristics, record staging sequences, monitor critical
environmental conditions and assure command destruct capability.
Generally, the information necessary to evaluate the instrumentation
required would be type of measurement desired, range, accuracy,
frequency response and resolution. Consideration must also be
given to environmental requirements, the type of ground data gathering
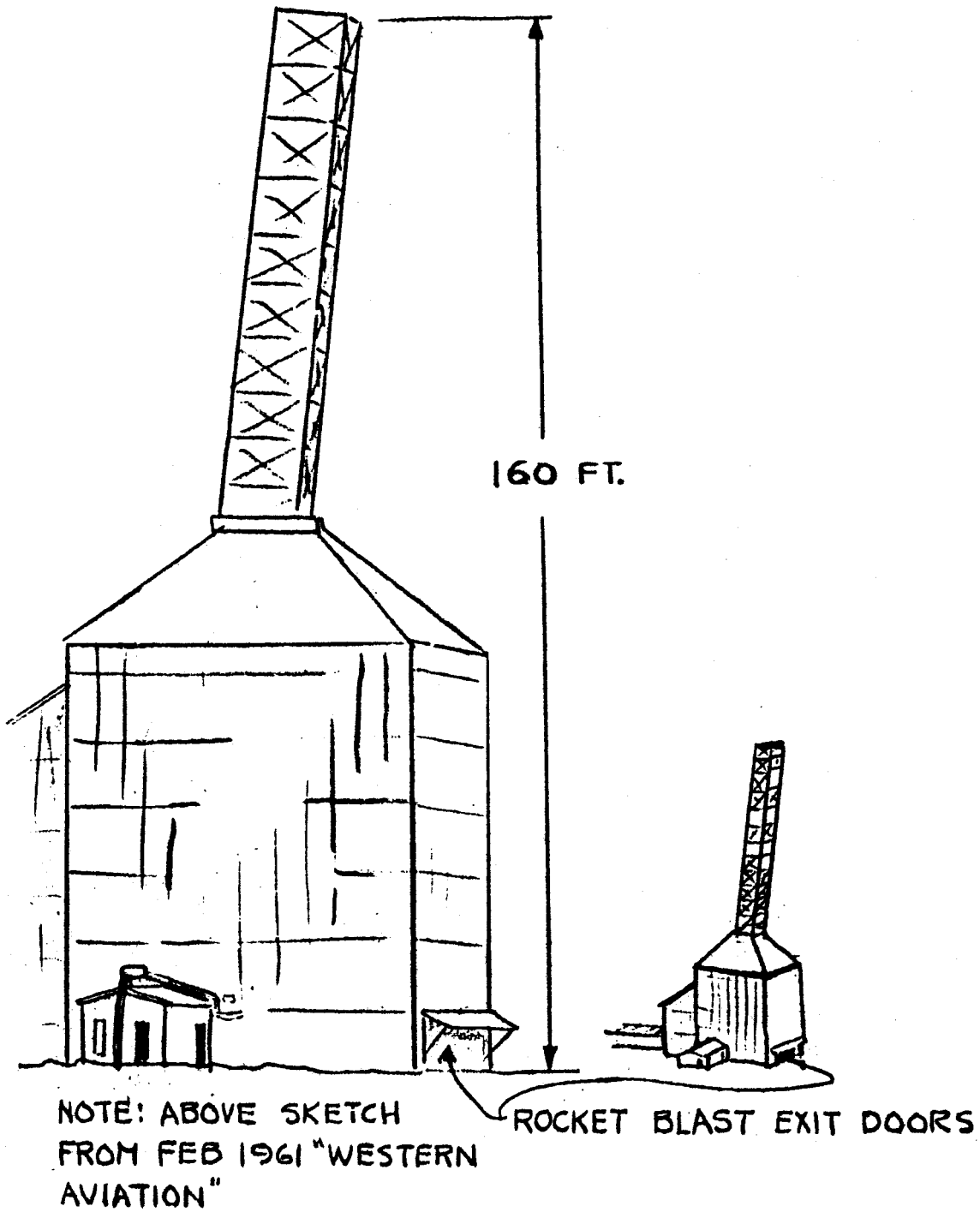equipment already available, and duration of operation.

160 FT.

NOTE: ABOVE SKETCH
FROM FEB 1961 "WESTERN
AVIATION"

ROCKET BLAST EXIT DOORS

FIGURE 26   WALLOPS ISLAND LAUNCHING TOWER

NOMENCLATURE

Symbol	Definition	Units
CL_a	Lift curve slope	per degree
C_D	Drag coefficient	
c. g.	Center of gravity from reference datum	in.
C. P.	Center of pressure from reference datum	in.
g	Gravitational acceleration	ft/sec^2
g_o	Gravitational acceleration at earth's surface*	ft/sec^2
g_s	Standard or normal gravitational acceleration	$32.174\ ft/sec^2$
G	Vibrational acceleration	ft/sec^2
I_{TOT}	Total impulse	lb-sec
$(I_{sp})AVG$	Average specific impulse, $\dfrac{I_{TOT}}{wc}$	sec
w_c	Total consumed weight	lb
w_p	Weight of propellant	lb
w_o	Weight of stage	lb
R_o	Earth radius*	ft
S	Aerodynamic reference area	ft^2
ΔV_{ID}	Ideal incremental velocity	ft/sec
α	Angle of attack	degrees
μ	Mass ratio, $\dfrac{w_o}{w_o - w_c}$	

* Where g_o and R_o represent conditions at a geodetic latitude of 35° on the International Ellipsoid of Reference:

$$g_o = 32.14389\ ft/sec^2 = 9.797459\ m/sec^2$$

$$R_o = 20,903,307\ feet = 6371.328\ kilometers$$

PAYLOAD DEFINITIONS

NET PAYLOAD: All weight not essential to the flying of the vehicle when the payload carrying stage is thrusting, but not including weight which is essential to the operation of a previous stage and which happens to remain attached to the payload carrying stage during its thrusting period.

GROSS PAYLOAD: All weight attached to the final payload carrying stage or rocket motor (tank weight shall be considered part of motor weight for liquid fuel systems).

GROSS EXPERIMENT PAYLOAD: Actual weight of instruments, batteries, telemeters, and associated bracketry that is carried while the payload carrying stage is thrusting.

VEHICLE STAGING DEFINITION
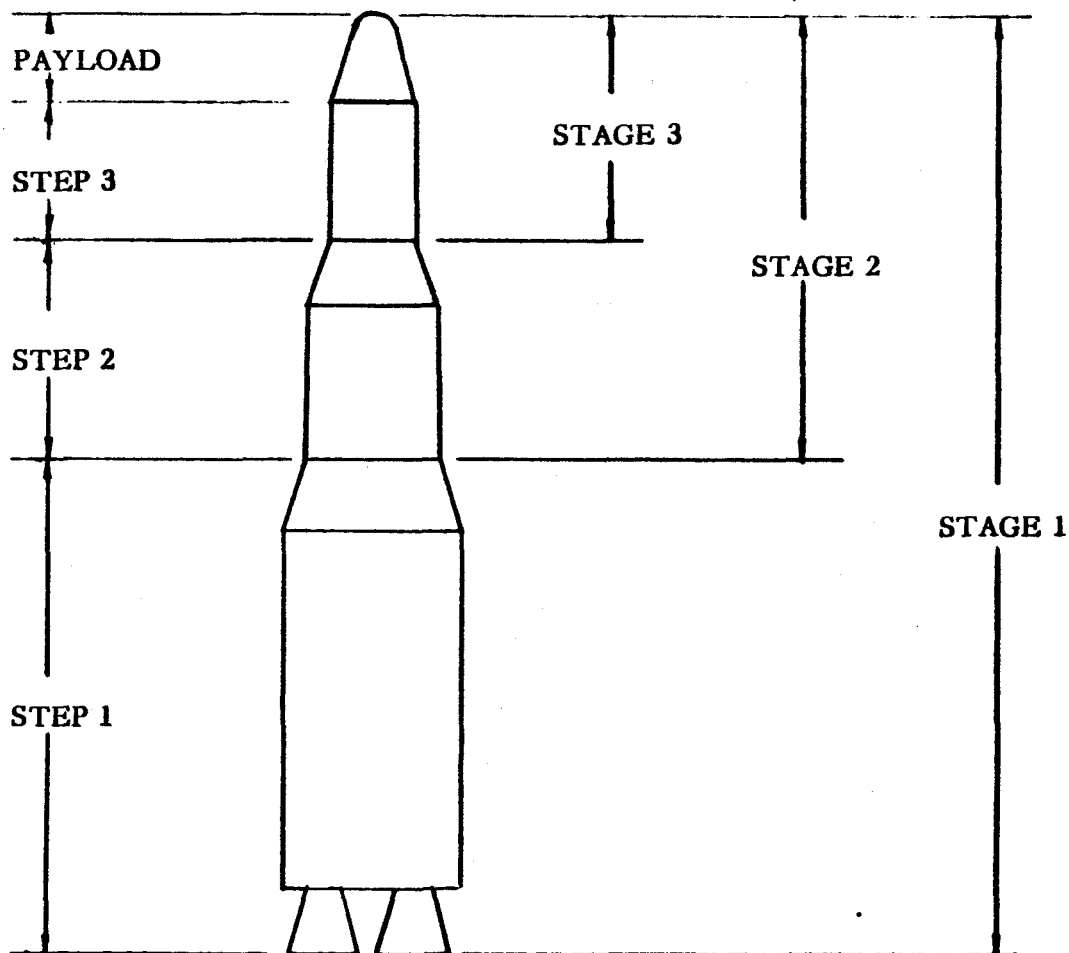
"Stage" is the preferred nomenclature when referring to system operation. "Step" is the preferred nomenclature when referring to the precise location or to the weight of a specific component.

REFERENCES

1. Grey, Jerry, "Aerodynamic Design and Performance Study of the Iris Sounding Rocket, Part 2, Performance Study and Final Design" Report No. ARC-12, Greyrad Corporation, Livingston, New Jersey, dated 15 March 1960.

2. "Handling Instructions for the Iris Sounding Rocket, 52 KS 3850, MARC-13A1, " Atlantic Research Corporation, undated.

3. Sargent, W. H., "Estimated Flight Performance, Iris 5.02", NASA, dated 30 October 1960.

4. Data Sheet, "Iris High Altitude Sounding Rocket, " Atlantic Research Corporation, undated.